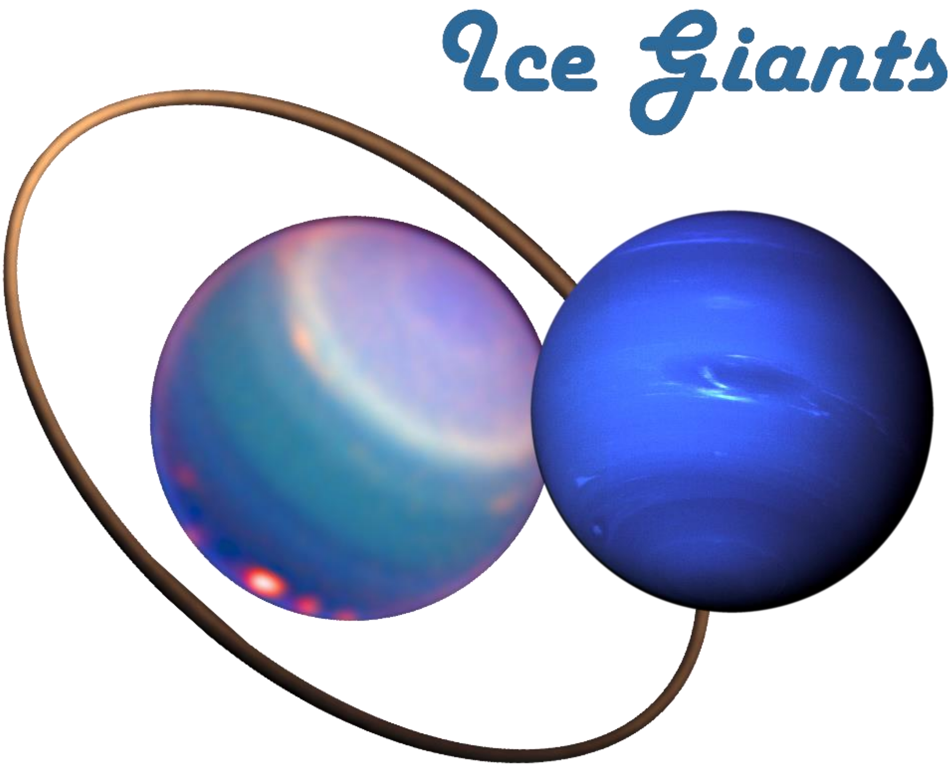
CDF Study Report
Ice Giants
A Mission to the Ice Giants – Neptune and Uranus



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FRONT COVER

Study Logo showing the two ice giants
Neptune and Uranus

STUDY TEAM

This study was performed in the ESTEC Concurrent Design Facility (CDF) by the following interdisciplinary team:

TEAM LEADER			
AOCS/GNC		POWER	
COMMUNICATIONS		PROGRAMMATICS/ AIV	
CONFIGURATION		CHEMICAL PROPULSION	
COST		RISK	
AEROTHERMO DYNAMICS		SIMULATION	
DATA HANDLING		MISSION ANALYSIS	
GS&OPS		STRUCTURES	
RADIATION/ ATMOSPHERIC MODEL		SYSTEMS	
MECHANISMS		THERMAL	

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1 INTRODUCTION

1.1 Background

A mission to the Ice Giants (Neptune and Uranus) will be among the ones examined by the next Planetary Sciences Decadal, which also fits with the potential launch opportunity, with a Jupiter swing-by, that would allow to reach both planets by launching in the early 2030s.

ESA is exploring potential contributions to a NASA-led mission to the ice giants aimed at understanding the interior structure and bulk composition of the planet(s) (including isotopes and noble gases).

ESA and NASA agreed to study a palette of possible configurations of varying cost to ESA and complexity, keeping in mind the need for clear interfaces.

It is important to keep this background in mind and remember that this study is not analysing a specific science proposal but trying to understand potential contributions following a top–down approach.

Requested by SCI-FM and funded by GSP, the M* (Ice Giants) study was set to analyse the feasibility of "stand-alone" elements provided by ESA to be part of the NASA-led mission to Uranus, Neptune and their moons *(M-class mission budget but not proposed following a Cosmic Vision Programme Call, hence M*).*

The study was carried out by an interdisciplinary team of experts from across ESA sites with the active participation of experts from NASA/JPL and the European science community (represented by the four members of the Science Study Team). The study consisted of 9 sessions, starting with a kick-off on the 7th November 2018 and ending with an Internal Final Presentation on the 12th December 2018.

The Mission

The potential mission contributions to be studied were:

- An individual spacecraft (orbiter), complementary to a NASA one. In this scenario the ESA orbiter would target one of the ice giants while the NASA spacecraft would fly to the other one.
- An atmospheric probe to either of the two planets, transported and released by a NASA orbiter.
- A lander to Triton (Neptune's largest Moon), transported and released by NASA orbiter.

The reference payload suites to be considered for the purpose of this study for the various elements were put together by the M* Ice Giants PL Team in liaison with representatives of the Scientific Community.

1.2 Objective

- The goal of the CDF study was to: Establish conceptual designs for the key European element(s) identified above in order to assess the mission feasibility identifying the required resources and defining the interfaces with the international partner

- Highlight the technological areas for which mission enabling developments would be required
- Define the programmatic approach and the schedule constraints for the studied option(s)
- Assess the mission cost for the studied option(s), taking into account that the ESA contribution shall fit within an M-class mission budget, i.e. 550 MEuro (excluding Member state contributions like Payload).

1.3 Scope

As previously stated, the scope of the study was not to analyse a specific science proposal but trying to understand potential contributions following a top–down approach. The defined study planning for the allocation of the associated sessions was based on the following assumptions:

- Orbiter
  - Design Target: Neptune
  - Design sensitivity analysis to Uranus
  - Payload: as specified by M* Ice Giants PL Team
  - Orbiter does not carry a probe
- Probe
  - Design Target: Neptune and Uranus
  - Reference: PEP CDF Study + Deltas Assessment (Designs very similar for Neptune and Uranus (small differences identified) – assumption based on PEP CDF Study)
  - Payload: as specified by M* Ice Giants PL Team/PEP PL complement; Design Assumption: PEP PL operating between 1 and 10 bars in 90 minutes (in PEP: free fall from 10 to 100 bars)
  - Released by NASA Orbiter
- Lander
  - Design Target: Triton
  - Payload: as specified by M* Ice Giants PL Team
  - Released by NASA Orbiter

1.4 Document Structure

The layout of this report of the study results can be seen in the Table of Contents. The Executive Summary chapter provides an overview of the study; details of each domain addressed in the study are contained in specific chapters.

Due to the different distribution requirements, cost information is removed from this version of the report.

2 EXECUTIVE SUMMARY

2.1 Study Flow

Requested by SCI-FM and funded by GSP, the M* (Ice Giants) study was setup to analyse the feasibility of "stand-alone" elements provided by ESA to be part of a NASA-led mission to Uranus, Neptune and their moons *(mission of opportunity, M-class mission budget but not proposed following a Cosmic Vision Programme Call, hence M*).*

The study was carried out by an interdisciplinary team of experts from across ESA sites with the active participation of experts from NASA/JPL and the European science community (represented by the four members of the Science Study Team). The study consisted of 9 sessions.

The study investigated:

- An individual spacecraft (orbiter) to either of the two planets;
- An atmospheric probe to either of the two planets, transported and released by a NASA orbiter;
- A preliminary sizing of a lander to Triton (Neptune's largest moon), transported and released by a NASA orbiter.

2.2 Neptune

2.2.1 Requirements and Design Drivers

The following science objectives and mission requirements were the starting point of the probe, orbiter and lander design.

2.2.1.1 Probe

Objectives:

- To determine the planet's bulk composition, including abundances and isotopes of heavy elements;
- To determine the compositional, thermal and dynamical structure of the atmosphere.

An atmospheric entry probe targeting the 10-bar level would yield insight into two broad themes: i) the formation history of the ice giants and, in a broader extent, that of the Solar System, and ii) the processes at play in planetary atmospheres.

The mission and system requirements of the probe are listed in the table below.

Mission Requirements		
Req. ID	**Statement**	**Parent ID**
MIS-010	The cost of the mission shall fit within a M-class mission	
MIS-020	The launch of the probe will be in the period of 2029-2034	

Table 2-1: Probe mission requirements

System Requirements		
Req. ID	**Statement**	**Parent ID**
SYS-010	The probe shall be carried by the NASA orbiter to Neptune	
SYS-020	The science observations of the probe shall occur during the descent from 1 bar to 10 bar and shall be 90 minutes	
SYS-030	The data generated on-board of the probe shall be transmitted to the orbiter in real time	
SYS-040	The orbiter shall serve as a relay for the probe during probe operations	
SYS-050	The probe shall perform a direct entry.	
SYS-060	The probe shall have Earth visibility during entry. *Note: to allow for UHF carrier monitoring of the probe from Earth during entry (see 36.5.2)*	

Table 2-2: Probe system requirements

2.2.1.2 Orbiter

The highest priority is the study of the interior structure of the planet. Secondary and equal priorities are listed below:

- Planetary dynamo
- Atmospheric temperature and dynamics
- Ring science
- Moons science (with a potential focus on Triton)
- Solar wind magnetosphere-ionosphere interactions.

The mission and system requirements of the orbiter are listed in the table below:

Mission Requirements		
Req. ID	**Statement**	**Parent ID**
MIS-010	The mission shall be launched within a timeframe of 2029 to 2034.	
MIS-020	The mission shall be compatible with launch via a SLS Block 1B.	
MIS-030	The total mission cost shall be within an M-class ESA science mission budget.	
MIS-040	The mission shall be capable of performing in-situ and observational measurements at Neptune with a payload suite defined by the Study Science Team.	
MIS-050	The mission shall be capable of performing in-situ and observational measurements at Triton during flybys, with a payload suite defined by the Study Science Team.	
MIS-060	The mission shall include at least do 2 years (TBC) of science operations at Neptune.	

Table 2-3: Orbiter mission requirements

System Requirements		
Req. ID	**Statement**	**Parent ID**
SYS-010	The orbiter shall be compatible with a dual launch with a NASA orbiter on an SLS Block 1B in a TBD configuration.	
SYS-020	The orbiter shall be delivered to NASA for final integration onto the launcher.	
SYS-030	The orbiter delivery date to NASA shall be compatible with the selected launch date and any pre-launch activities agreed between ESA and NASA.	
SYS-040	The orbiter shall be compatible with a storage of TBD months before integration onto the launcher.	
SYS-050	The orbiter design shall allow late access for integration of the RTGs under the launcher fairing.	
SYS-060	The orbiter shall include a payload suite of 116 kg (TBC).	
SYS-070	The operational lifetime of the orbiter shall be at least 15.5 years after launch.	
SYS-080	[DELETED]	
SYS-090	The orbiter shall rely solely on its own power source(s) during cruise with the NASA orbiter.	
SYS-100	The orbiter shall provide a SpaceWire interface to the NASA orbiter.	
SYS-110	The orbiter shall be asleep during cruise with the NASA orbiter, apart from periodic checkouts.	
SYS-120	The orbiter shall not require any active thermal control from the NASA orbiter.	
SYS-130	The orbiter shall separate from the NASA orbiter before the Jupiter swing-by.	
SYS-140	The orbiter shall be able to perform an independent interplanetary transfer from separation until Neptune.	
SYS-150	The orbiter shall be able to insert into orbit around Neptune.	
SYS-160	The orbiter shall be able to download all gathered science data within the nominal mission duration (TBC).	
SYS-170	The orbiter shall be compatible with all environments from integration until EOM.	
SYS-180	The orbiter shall include redundancy for all mission-critical functionalities (TBC).	

Table 2-4: Orbiter system requirements

2.2.1.3 Lander

Objectives:

- Map surface geology at the landing site
- In situ surface and subsurface characterisation
- Determine surface composition, including organics; search for variations evidence for mass exchange/volatile transportation

- Determine the composition of Triton's atmosphere
- Investigate moon-magnetosphere interactions

The main design drivers for the Triton lander include:

- Release strategy: from orbit around Triton or during flyby only. This has a strong impact on the delta V
- Low atmospheric density, implying that a propulsion-only descent and landing is assumed
- Need for throttled / pulsed propulsion capabilities in a closed-loop GNC system for the final descent manoeuvre (technology gap)
- Possible need of reconnaissance imaginary created by another mission to enable high level selection of safety areas
- Instruments/science (during descent and surface operations): Mass / power / data / temperature/ Operations timeline
- Available communications window(s) duration.

The high-level mission requirements of the Triton lander are listed in the table below:

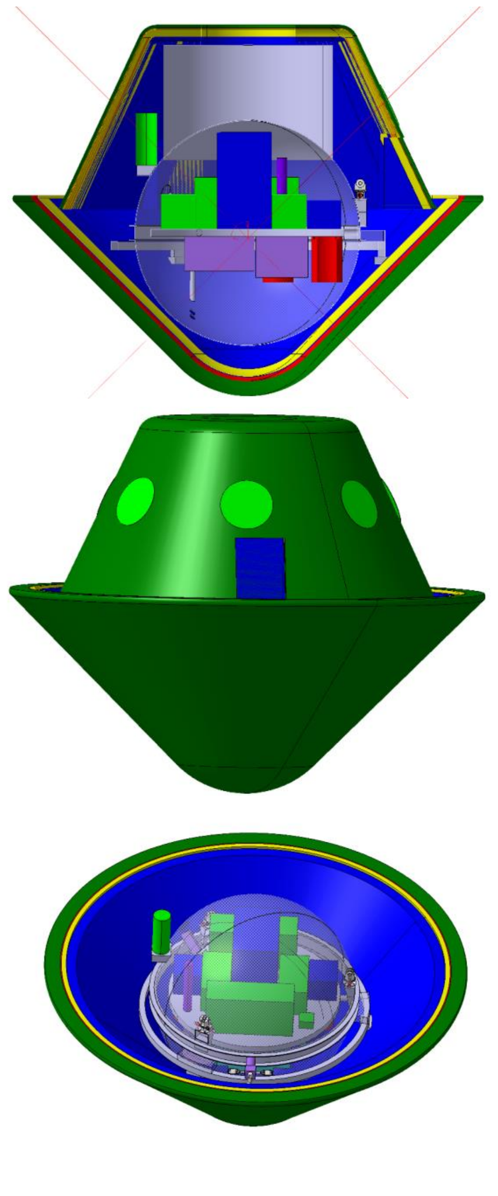
	Main Requirements	
Req. ID	**Statement**	**Parent ID**
MI-010	The Triton Lander shall land a payload of 11.18 kg	
MI-020	The Triton Lander shall be released from Triton fly-by	
MI-030	The Triton Lander shall perform a soft landing manoeuvre of 4637 m/s	
MI-040	The Triton Lander shall operate during one week of lifetime	

Table 2-5: Triton Lander mission requirements

2.2.2 Mission

2.2.2.1 Probe baseline design

Probe		
Mass (Incl 20% system margin)	Mass w/o TPS&TC: 191 kg TPS&TC: 151 kg Mass with TPS: 342 kg	
Ballistic coefficient	Projected area: 1.43 m² (diameter 1.35 m) Front shield area: 1.99 m² Cd: 1.07 BC: 228 kg/m²	

Probe		
Payload	• Atmospheric Structure Instrument • Camera-Radiometer • Mass Spectrometer • Photometer • USO-Doppler	
EDS	2 subsonic parachutes: pilot (M=0.8) and main	
TPS	Front shield: 51.9 mm thickness, 129 kg Back shield: 31.4 mm, 19.9 kg	
GNC	2 redundant IMUs 2 parachute deployment switches	
Mechanisms	Back and Front Shell Separation Mechanisms Parachute Swivel Mechanism Mortar parachute pyro cutter Spin Eject Mechanism [Probe side]	
Communications	UHF redundant chain Patch antenna on the backshell Helix antenna during descent	
Power	4 x 3 kg batteries PCDU	
Data Handling	CDMU including timer	
Structures	61.3 kg of structures (of which 28.5kg are in the descent module)	
Thermal	31 RHUs, MLIs, Front shield radiator, pressure vessel insulation	

2.2.2.2 Orbiter baseline design

Orbiter	
Mass (Incl 20% system margin)	Dry mass: 1605 kg Propellant mass (excl. margin): 1991 kg Wet mass: 3969 kg
Payload	Camera Imaging Spectrometer Ion and Neutral Mass Spectrometer Magnetometer

Orbiter	
	Macrowave radiometer Ultra Stable Oscillator (USO) Ka-band transponder
Propulsion	2x main bipropellant thrusters (1000 N) 16x RCS thrusters (10 N) 3x pressurant tanks (2x 120 L and 1x 66 L tanks) 4x propellant tanks (550 L)
AOGNC	1x coarse rate sensor 2x navigation cameras 2x IMUs 2x star trackers 4x reaction wheels (+ RCS thrusters)
Communications	X-band uplink/downlink Ka-band downlink (42 kbps) Science volume downlinked: 0.48 Gb/day Communication window duration: 3.2 h/day Data volume generated by EOM: 350 Gb
Power	3x eMMRTGs (EOM Power = 90W) 4x 48kg batteries
Data Handling	Redundant OBC + 1Tbit of storage
Structures	303 kg
Thermal	Heaters + use of the eMMRTG thermal dissipation

On the orbiter, trade-offs were performed, including to investigate the dual launch configuration on the SLS and the number of RTG's used on the orbiter.

2.2.2.3 Triton lander baseline design

The lander design was based on a draft payload definition which was reduced in mass and power consumption to be able to be accommodated on a small lander. Based on the initial projected payload mass of 11.18 kg, a lander of >2000kg was estimated for landing from Triton flyby.

An alternative top-down assessment was then performed. Using a lander wet mass of 350 kg, an estimated 1.5 kg was predicted as available from Triton flyby. This is still significantly less than the reduced model payload of 2.24 kg.

It was noted that an option to release the lander from Triton orbit (rather than flyby) would significantly reduce the delta-v required for a soft landing, and thus increase the payload mass / lander wet mass ratio.

2.3 Uranus

The requirements and baseline designs for the Uranus case are the same as for the Neptune case, with the exception of the planetary destination and these other changes highlighted below.

2.3.1.1 Probe

The probe design of the Uranus case was kept the same as for Neptune. The only change is the atmospheric entry and descent trajectory due to the different spin properties of Uranus.

2.3.1.2 Orbiter

Given the shorter interplanetary transfer time to Uranus, the lifetime requirement (SYS-070) would allow reduction, as highlighted in the table below. Nonetheless, the baseline lifetime as used for the Neptune case was maintained, in order to allow as much re-use of the Neptune design as possible. For this reason, the baseline science operations phase (SYS-080) was extended to 4 years in order to give an equivalent total mission duration as for the Neptune case.

Orbiter System Requirements		
Req. ID	**Statement**	**Parent ID**
SYS-070	The operational lifetime of the orbiter shall be at least 13.5 years after launch (baseline: 15.5 years)	
SYS-080	The science operations phase of the mission shall be at least 2 years (baseline: 4 years)	

Table 2-6: Uranus Orbiter system requirements

Given that the study goals foresaw reusing the Neptune design as much as possible for the Uranus case, it was noted that some subsystems may be oversized. The baseline is nonetheless summarised in the following table.

Orbiter	
Mass (Incl 20% system margin)	Dry mass: 1914 kg Propellant mass (excl. margin): 2484 kg Wet mass: 4398 kg
Payload	Camera Imaging Spectrometer Ion and Neutral Mass Spectrometer Magnetometer Microwave radiometer Ultra Stable Oscillator (USO) Ka-band transponder
Propulsion	1x main bipropellant thruster (1000 N) 16x RCS thrusters (10 N)

	3x pressurant tanks (120 L) 4x propellant tanks (550 L)
AOGNC	1x coarse rate sensor 2x navigation cameras 2x IMUs 2x star trackers 4x reaction wheels (+ RCS thrusters)
Communications	X-band uplink/downlink Ka-band downlink (94 kbps) Science volume downlinked: 1.09 Gb/day Communication window duration: 3.2 h/day Data volume generated by EOM: 1.6 Tb
Power	3x eMMRTGs (EOM Power = 90W for 4 year science phase) 4x 48kg batteries
Data Handling	Redundant OBC + 1Tbit of storage
Structures	303 kg
Thermal	Heaters + use of the eMMRTG thermal dissipation

Table 2-7: Orbiter system baseline (Uranus)

2.4 Technical Conclusions and Options

The M* Ice Giants study objectives were successfully achieved.

The major study findings for the probe, orbiter, and Triton lander are described below.

2.4.1 Probe

The PEP CDF Study was taken as the initial reference for the probe assessment, with only deltas with regards to PEP being assessed in the M* Ice Giants CDF Study. The most significant changes were:

- TPS mass: due to revised characterisation of the TPS material properties and an increase in size, the total mass of the TPS increased by 50% (despite a marginally lower entry velocity).
- Pressure range: scientific observations were changed to spend more time at lower pressures (1-10 bar), and as such the pressure vessel could be reduced. However the increased observation time at these pressures resulted in a larger main parachute.
- A mass reduction on the DHS was achieved via the latest technology developments.
- 31 RHUs were installed to survive the 20-day coast phase.

2.4.2 Orbiter

- For the Neptune design case, it was demonstrated that downlink of the generated science data versus the available energy would be a significant challenge. However this could be revisited with an revised operations concept and further iterations on the orbital timeline at Neptune (including moon tours). The inclusion of a larger high gain antenna would also improve the available downlink (dependent on launch configuration and fairing size).
- For the Uranus case, the data volume constraints seemed more relaxed than with regards to Neptune.
- The availability and inclusion of 3 eMMRTGs was shown as essential to enable any type of useful science at the destination planet, even for the Uranus case.
- The EOM power of the RTGs must still to confirmed, and would have a large impact on the available energy for downlink and on the payload duty cycle.
- In a dual-orbiter scenario the availability of a combined 6+ eMMRTGs for both missions, and the implications of this on AIV, storage and launcher fairing access, would pose challenges.
- The trajectory used to target Uranus (and in particular, the flyby at Jupiter) might impose stringent requirements for radiation tolerance of up to 155 krad for all units.
- Technology developments shall be compatible with the programmatic requirement of TRL 6 by end of 2022 (corresponding milestone: mission adoption) for a launch on 13 February 2031.

2.4.3 Lander

A rough scaling exercise from an existing lander study was performed, in order to derive a quick relationship to estimate available payload masses. It was noted that the relationship is optimistic for higher wet masses, and that landing from Triton orbit would significantly reduce the propellant mass required.

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3 SCIENCE OBJECTIVES

3.1 Background

The ice giants, Uranus and Neptune, have been visited by the Voyager 2 spacecraft in 1986 and 1989, respectively. These two fantastic flybys raised some questions that still need to be answered by dedicated missions. The ice giant system is a distinct class of planets, fundamentally different from the better explored gas giants, Jupiter and Saturn. Their study is critical and absolutely necessary to advance our understanding of the solar system origin and evolution RD[1] to RD[6]. As ice giant type planets represent around ¼ of exoplanet population, they are the only laboratory in which one can perform in-situ experiments to understand exoplanet formation, dynamos, systems and magnetospheres RD[2]. The moon system of Uranus and Neptune is also extremely interesting to explore. In particular, the Triton moon is very likely a captured Kuiper Belt object RD[3], and is predicted to harbour a subsurface ocean. The choice between which system to explore is not straightforward. Uranus and Neptune are equally important, but are different from each other.

3.2 Mission Justification

A mission to the icy giants will be among the ones examined by the next Planetary Sciences Decadal Survey RD[1]. Given the broad science goals, the two planets to explore, and the different mission elements under consideration, there is a clear opportunity to collaborate with NASA, similarly to the international Cassini-Huygens mission.

There is a large scientific community behind such planetary missions [RD[2], RD[3], RD[4], RD[5], RD[6]].

A launch opportunity has been identified in 2031, which would allow reaching both planets with one single launch.

It’s time to explore Uranus and Neptune again!

3.3 Science Objectives

The science objectives are largely taken from RD[1]. Since three mission elements have been analysed in this CDF study, it was decided to define one science traceability matrix per element, which can be found in annex A. The following subsections list the science objectives per element (by alphabetical order). The model payload to address these objectives are discussed in the instrument section.

3.3.1 Atmospheric Probes

Regarding the probes, the highest priority is to determine the planet’s bulk composition, including abundances and isotopes of heavy elements, while a second priority is the determination of the compositional, thermal and dynamical structure of the atmosphere. An atmospheric entry probe targeting the 10-bar level would yield insight into two broad themes: i) the formation history of the ice giants and, in a broader extent, that of the Solar System, and ii) the processes at play in planetary atmospheres.

3.3.2 Orbiters

The highest priority is the study of the interior structure of the planet. Secondary and equal priorities are listed below:

- Planetary dynamo
- Atmospheric temperature and dynamics
- Ring science
- Moons science
- Triton (in the case of the Neptune orbiter)
- Solar wind magnetosphere-ionosphere interactions.

3.3.3 Triton Lander

The science objectives were discussed in the context of an orbiter (with the scientific objectives relevant to Triton).

There are three groups of decreasing priorities:

Priority #1 (highest):

- Map surface geology at the landing site
- In situ surface and subsurface characterisation.

Priority #2:

- Determine surface composition, including organics; search for variations evidence for mass exchange/volatile transportation
- Determine the composition of Triton's atmosphere.

Priority #3:

- Investigate moon-magnetosphere interactions.

3.4 Mission Requirements

The main requirements are:

- Atmospheric probes:
  - Payload model recommended by the SST and same payload for both planets
  - Measurements to be performed in the 1-10 bars range, and for a duration of 90 minutes. It is expected that measurements of atmospheric structure will start in fact above 1 bar level (during entry). As in the case of the Galileo probe, a lower altitude could be reached, even with a design for 10 bars.
  - Visibility from Earth of the Entry and Descent phases is desired to track the probe's carrier signal (as done for Huygens and ExoMars2016-Schiaparelli)
  - Direct entry
  - Data transmitted in real time to the NASA Orbiter, which serves as relay to Earth
- Orbiters:
  - Payload model recommended by the SST and same payload for both planets

- Similar trajectories than in [RD1] around the ice giants.
- Launched in stacked configuration with the NASA orbiter (SLS launch assumed)
- Science operations duration: at least 2 years.

- Triton Lander:
  - Payload model recommended by the SST.
  - No specific requirements for the landing site.

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NEPTUNE

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4 NEPTUNE MISSION ANALYSIS

The Mission Analysis work is based on the Dual Spacecraft, Single Launch scenario from Appendix A6 in RD[1], assuming launch in February 2031 on a SLS-IB heavy lift launch vehicle. Launch sends a composite (stack) of the Uranus and the Neptune orbiters directly to Jupiter together with a SEP stage. The Neptune orbiter separates on the transfer to Jupiter.

The Uranus and Neptune orbiters perform independent Jupiter swingbys in December 2032. The Uranus orbiter performs a very close swingby at a perijove altitude of 10,000 km, while the Neptune spacecraft a much higher one at around 857,000 km (see Figure 4-2), after which the two satellites travel in completely different directions, reaching their targets in April 2042 and September 2044, respectively.

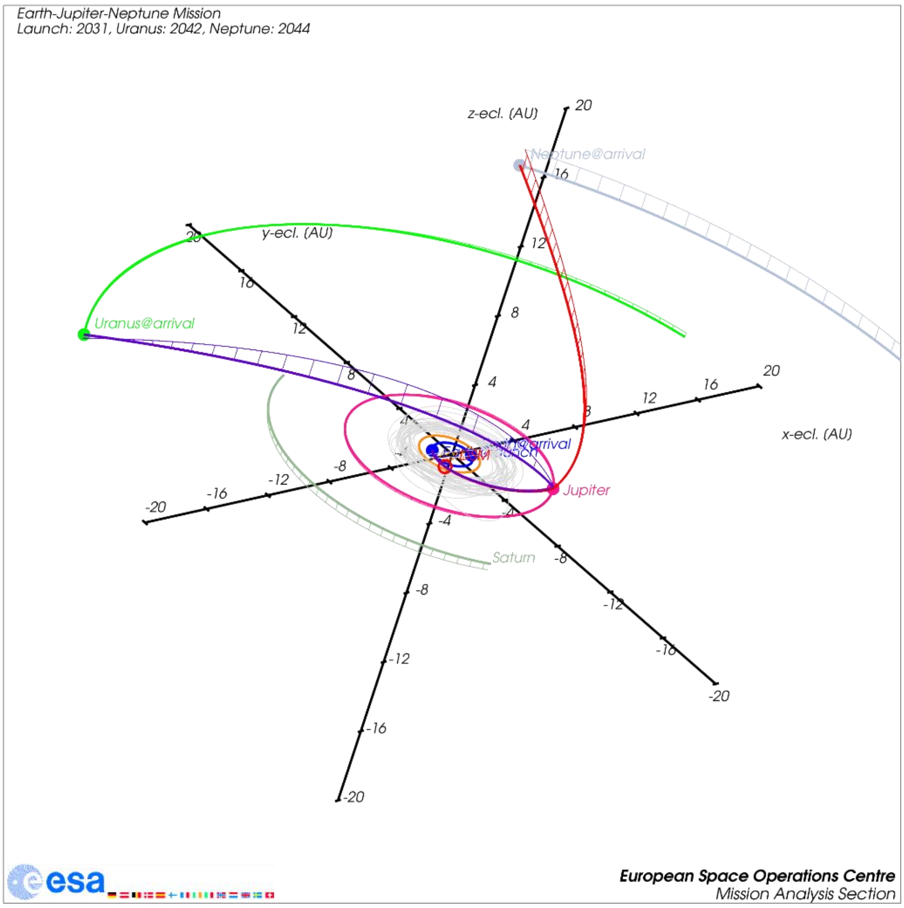

Figure 4-1: Dual Spacecraft, Single Launch Transfer Overview

The obtained arrival conditions at Uranus and Neptune (considered here) are the main input for all further analysis. The transfer scenario, timeline and arrival conditions would be significantly different for different assumptions on the overall mission.

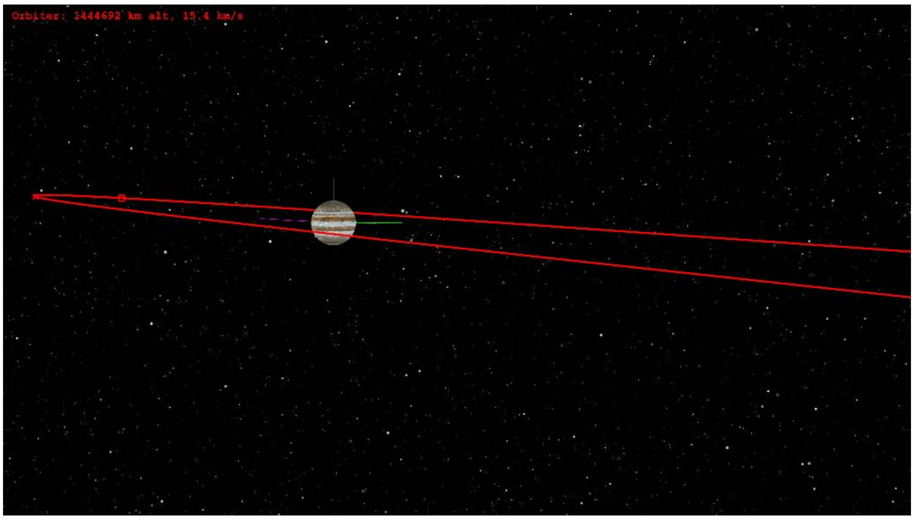

Figure 4-2: December 2032 Jupiter Swingby of the Neptune Mission

4.1 Atmospheric Probe

4.1.1 Requirements and Design Drivers

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
MA-010	Consistency with the entry conditions assumed in the earlier PEP study RD[7], specifically, a FPA relative to the rotating frame of -35 deg at an EIP altitude of 600 km above the 1 bar radius	
MA-020	The atmospheric part of the probe mission shall take place during local daylight and with visibility from the Earth.	
MA-030	The atmospheric phase of the probe mission shall last up to 90 minutes	

4.1.2 Assumptions and Trade-Offs

Assumptions	
1	If ESA provides a probe this is assumed to be carried by a NASA-provided orbiter. *Note: Any mention of the orbiter in this chapter is discussing the NASA orbiter. The design of the ESA orbiter has a dedicated chapter.*
2	The probe is assumed to be deployed such that it enters the Neptune atmosphere at a location close to the equator and with a prograde velocity orientation. This is not consistent with a Neptune tour that is optimised for observation of the main moon Triton, which is on a retrograde, circular orbit, inclined by 157 deg wrt Neptune's equator plane.

4.1.3 Baseline Design

The Neptune entry diagram for the given scenario is displayed in Figure 4-3. Neptune's equator is inclined by 28.32 deg with respect to its orbit. In the given case, the Sun and Earth direction and the direction towards the incoming probe are all close to the equator and close to the noon meridian. The Sun and Earth visibility terminators are shown. All entry points above the Earth visibility terminator have Earth visibility at entry. The steeper the entry, the better are the Earth visibility conditions at entry, and consequently, the longer the time after entry before Earth loses visibility of the entry probe.

Entry locations and directions of flight are shown for inertial flight path angles of -25, -35 and -45 deg. For -25 deg, all entry points are either beyond the Sun and Earth visibility terminator or close to it, so a daylight mission with Earth visibility would be impossible. Prograde entry at 0 deg of latitude minimises the relative entry velocity to 23.2 km/s, compared to over 27 km/s for retrograde, equatorial entry.

An inertial entry flight path angle of -35 deg appears to allow missions that are consistent with the Sun and Earth visibility requirement, but only if entry is prograde. In the present study a relative entry flight path angle of -35 deg is required, which translates into some offset in the inertial FPA. However, Figure 4-3 is still qualitatively applicable.

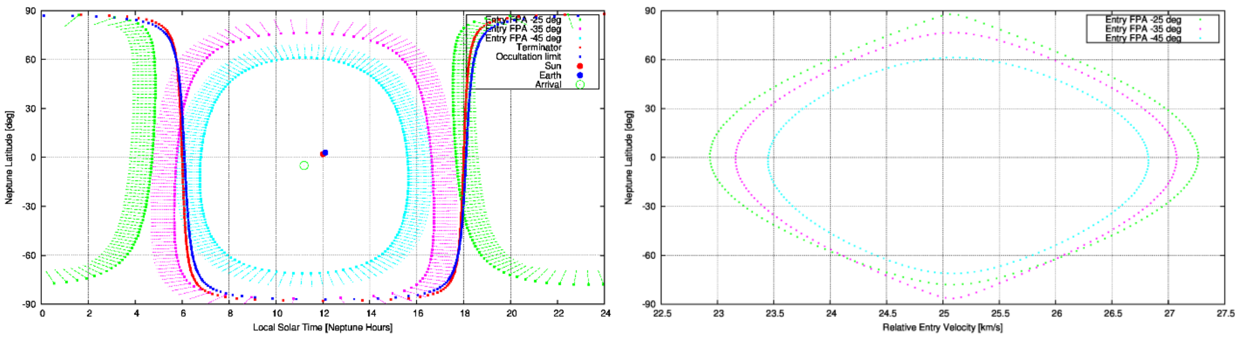

Figure 4-3: Entry Plot and EIP Velocities for 2044/9/1 Arrival at Neptune

4.1.4 Budgets

Table 4-1 lists the entry conditions for prograde, equatorial entry with a relative FPA of -35 deg at an EIP altitude of 600 km above the 1 bar radius. All data are given in the planet-centered rotating frame.

Altitude [km]	600.133
Velocity [km/s]	23.082
FPA [deg]	-35.039
Longitude [deg E]	-8.821
Latitude [deg N]	-0.749
Azimuth [deg]	84.468

Table 4-1: Entry Conditions for Prograde, Equatorial Entry at -35 deg Relative FPA

Note that the longitude value given here applies only to entry at the stated epoch. The entry longitude can be modified at negligible delta-v cost just by changing the arrival time by +/- 8 hours, which will not affect any of the other parameters. Conversely, the entry latitude can be changed only by applying a steeper or shallower entry FPA or by a significant change in the arrival date, all of which would have a significant effect throughout the mission design.

Even if there is Earth visibility, an array of terrestrial radio telescopes will at best only be able to capture the carrier signal. Data transmission will have to take place via the orbiter, which will be performing NOI while the probe is performing atmospheric entry. The determining parameter for the coverage quality is the periapsis altitude of the orbiter.

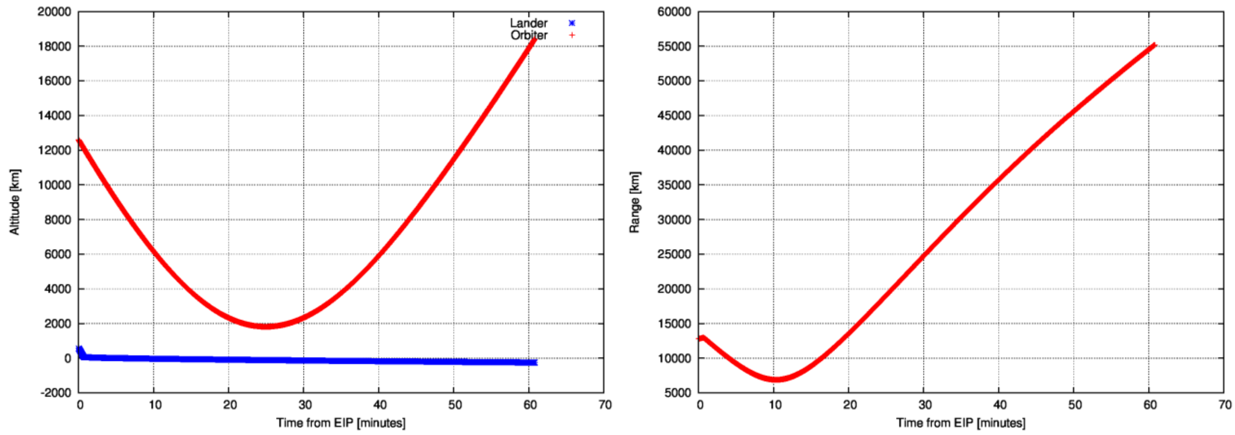

Figure 4-4: Altitudes and Slant Range, Target Periposeidon Alt. 2000 km

For a targeted periposeidon altitude of 2000 km above the 1 bar radius the altitude of probe and orbiter and the probe-orbiter slant range are shown in Figure 4-4, while Figure 4-5 gives the evolution of Earth aspect angle (EAA) and Orbiter aspect angle (OAA), i.e., the angle between the symmetry axis of the entry probe and the directions to Earth or orbiter. The probe symmetry axis is assumed to be aligned in the opposite direction of the current relative velocity wrt. the rotating Neptune atmosphere.

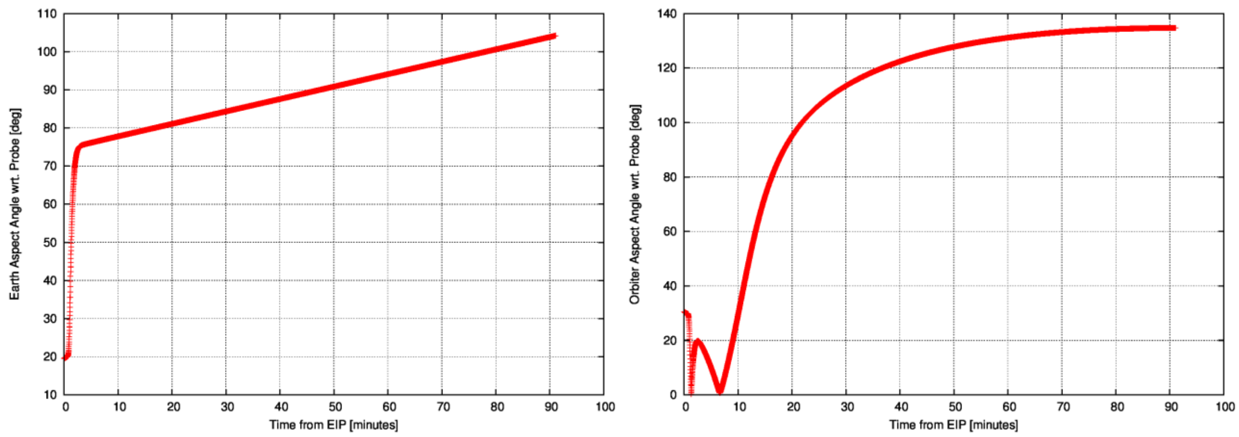

Figure 4-5: EAA and OAA, Target Periposeidon Altitude 2000 km

Following entry, the probe will slow down through aerodynamic drag and its relative flight path angle will quickly deepen from -35 deg to near-vertical. As a consequence, the EAA undergoes a strong initial increase, followed by a slow drift caused by the probe being carried along by the rotating planet.

Conversely, the OAA initially approaches zero as the orbiter, which was trailing the probe on a higher and slower orbit, catches up and passes the probe directly above. Around 10 minutes after entry, the OAA increases fast as the orbiter races ahead while the probe is moving only slowly with respect to the atmosphere and its lateral motion is due only to the rotation of the latent. The OAA goes above 90 deg around 19 minutes after entry. Depending on the opening angle of the probe antenna pattern the orbiter will lose contact at the latest at that point, or likely some time earlier. A target orbiter periposeidon altitude of 2000 km is inconsistent with a probe mission duration of 90 minutes. (Note that for a retrograde entry, the descending probe would be carried in the opposite direction of the orbiter flight by the rotation of the planet, so the OAA would rise faster and an even higher target periposeidon altitude would be required to ensure relay coverage for a 90 minute probe mission.)

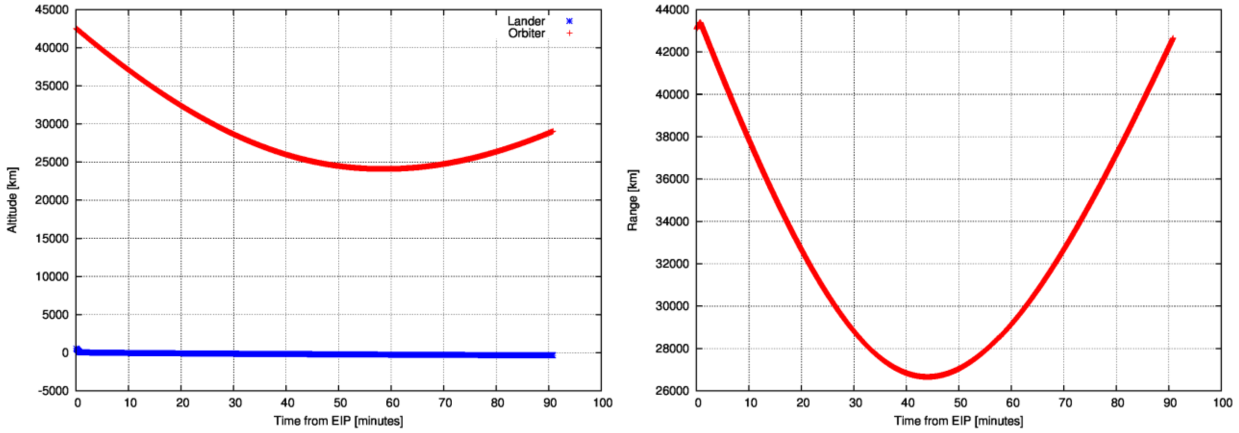

Figure 4-6: Altitudes and Slant Range, Target Periposeidon Alt. 25000 km

The same set of diagrams has been produced assuming a target periposeidon altitude of 25000 km. In this case, the orbiter arrival is delayed significantly and the probe-orbiter geometry is much different, leading to a much larger slant range with a minimum of 26500 km but also a time of 90 minutes from entry to the point where the OAA reaches 77 deg. This indicates that a target periposeidon altitude of around 25000 km is required to support a 90 minute probe mission.

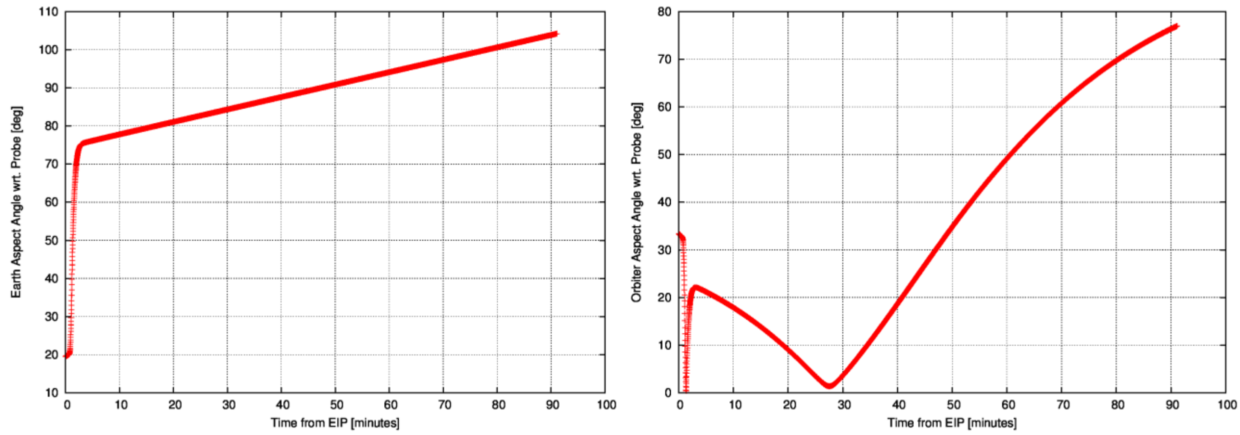

Figure 4-7: EAA and OAA, Target Periposeidon Altitude 25000 km

Note that the switch to a high periposeidon altitude has implications not only on the ODM and NOI size but that it may also lead to intersection of the rings. This must be studied in detail.

4.2 Orbiter

4.2.1 Assumptions and Trade-Offs

	Assumptions
1	In the present study, a target periposeidon altitude of 2000 km above the 1 bar radius has been assumed.
2	For the ESA-provided orbiter, the communications with the entry probe, which will then not be provided by ESA, is not assumed to be object of the study
3	The tour design is assumed not to be object of the study. The information related to the tour contained in the NASA document RD[1] is considered to be applicable.

4.2.2 Baseline Design

For the orbiter study, no considerations related to deploying a Neptune entry probe have been taken into account. Therefore, the ODM is not budgeted. The NOI manoeuvre has been modelled for different T/M ratios, assuming different values of the target apoposeidon radius.

4.2.3 Budgets

Table 4-2 shows the NOI size and duration obtained via numerical propagation of the trajectory, assuming that the thrust acceleration is anti-tangential to the current poseidocentric velocity. The results are given for various values of the thrust/to mass ratio *at the start of the manoeuvre*, and for different target apoposeidon radii. The results are applicable independently of the inclination of the obtained orbit.

This flyby altitude would be achieved if no manoeuvre took place; it would also be the periposeidon altitude of the capture orbit if the manoeuvre were near-impulsive. The longer the manoeuvre duration, the more the osculating periposeidon is lowered during the burn. This, together with the significant gravity losses, should be taken into account when designing the propulsion system.

In the studied range of apoposeidon radius values, the impact on the NOI size is minimal. Only for much lower apoposeidon values will there be a marked increase in the NOI. This imposes constraints on the accuracy of the execution of the insertion manoeuvre, as any mis-performance would lead to a significant deviation of the obtained from the planned orbit.

All further details on the tour timeline and manoeuvre sequence are beyond the scope of the CDF study and should be taken from RD[1].

Description:	**Target periposeidon altitude 2000 km, apoposeidon radius 275 RN**	
Thrust/Mass ratio [N/kg]	**NOI [m/s]**	**Duration [s]**
0.25	2249	6429
0.5	2061	3025
0.75	2005	1977
1.0	1981	1470
Description:	**Target periposeidon altitude 2000 km, T/M Ratio 0.5 N/kg**	
Apoposeidon radius [RN]	**NOI [m/s]**	**Duration [s]**
275	2061	3025
250	2065	3030
225	2072	3036
200	2079	3044
	Target periposeidon altitude 25000 km, T/M Ratio 0.5 N/kg	
275	2615	3553

Table 4-2: NOI Size as Function of Various Parameters

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5 NEPTUNE SYSTEMS

5.1 Atmospheric Probe

5.1.1 Mission & System Requirements and Design Drivers

Mission Requirements		
Req. ID	**Statement**	**Parent ID**
MIS-010	The cost of the mission shall fit within a M-class mission	
MIS-020	The launch of the probe will be in the period of 2029-2034	

Table 5-1: Mission requirements

System Requirements		
Req. ID	**Statement**	**Parent ID**
SYS-010	The probe shall be carried by the NASA orbiter to Neptune	
SYS-020	The science observations of the probe shall occur during the descent from 1 bar to 10 bar and shall be 90 minutes	
SYS-030	The data generated on-board of the probe shall be transmitted to the orbiter in real time	
SYS-040	The orbiter shall serve as a relay for the probe during probe operations	
SYS-050	The probe shall perform a direct entry.	
SYS-060	The probe shall have Earth visibility during entry. *Note: to allow for UHF carrier monitoring of the probe from Earth during entry (see 36.5.2)*	

Table 5-2: System requirements

5.1.2 System Assumptions and Trade-Offs

Assumptions	
1	The PEP design is the reference for the Ice Giants probe (RD[7]).
2	The science payload of the probe is the same as for the PEP study (RD[7]).
3	The launch date will be 13/02/2031.
4	The RHUs will be provided by NASA.
5	The probe batteries can be charged and topped off before probe release.
6	The NASA orbiter can provide up to two hours of data relay. *Note: This implies an orbiter with a higher periapsis altitude (>25000 km) than assessed in the NASA Ice Giants study (RD[1]) as shown in the mission analysis chapter(4.2) .*
7	The TPS of the probe also shields the equipment inside from the radiation during Jupiter fly-by.

5.1.3 Mission System Architecture

5.1.3.1 Mission options

5.1.3.1.1 Coasting duration

The coasting duration for the PEP probe was assumed to be 20 days. An option to increase this to 60 days for Ice Giants was investigated, based on similar assumptions in RD[1]. During the coast phase of PEP it was also assumed that the probe would send sporadic telemetry (namely housekeeping and GNC data) to the orbiter. This required frequent activation of the probe units, and amounted to a total energy consumption of 297 Wh. These assumptions were also traded against a coastal phase design where only a timer (Mission Timer Unit, MTU) was operational, with all other units in hibernation. An updated MTU power assumption was used, requiring only 180 mW compared to the 272 mW timer of PEP.

Table 5-3 provides a summary of the trade-offs. The criteria considered are:

- Probe battery size (compared to the PEP battery of 11 kg)
- Orbit Deflection Manoeuvre (ODM) required by orbiter after probe release
- Thermal impact (preliminary assessment only).

	PEP-like operations + MTU		**“Only-MTU-on” case, smaller MTUs**	
System impacts	**20 days**	**60 days**	**20 days**	**60 days**
Total energy for coast phase (Wh)	297	891	86.4	259
Increase of PEP battery size (%)	-	+64.8%	-23.0%	-4.15%
ODM (m/s)	6.9	2.3	6.9	2.3
Estimated temperatures at arrival (without RHUs) (°C)	*-28*	*-136*	*Unknown (worse than PEP)*	*Unknown (worse than PEP)*

Table 5-3: Trade-off between the 20 day and 60 day cruise mode including or excluding the PEP like operations

This trade-off used the following assumptions:

- The ODM delta-v analysis assumed a target periposeidon altitude of 2000 km from the NASA report (note that a 20,000 km design case was shown to not affect the delta-v markedly).
- The lower temperature limit for all internal units in PEP was assumed to be -40°C.

- For the 60 days thermal case the unit temperatures are extrapolated linearly after the 20 days that were analysed in PEP.

Based on the table above it has been decided to go for the 20 days coast duration with only the MTU turned "ON". The impact on the delta-v budget of the orbiter is negligible and the benefits on the battery sizing and arrival temperature made this the obvious choice.

5.1.3.1.2 Entry flight path angle

The entry flight path angle (FPA) at an interface point of 600 km altitude needed to be selected such that:

- The entry will be prograde, in order to reduce the entry velocity
- In daylight (Copy from PEP but not really required)
- Visible from earth during the entry.

These entry condition requirements will drive the FPA of the probe. See Figure 5-1.

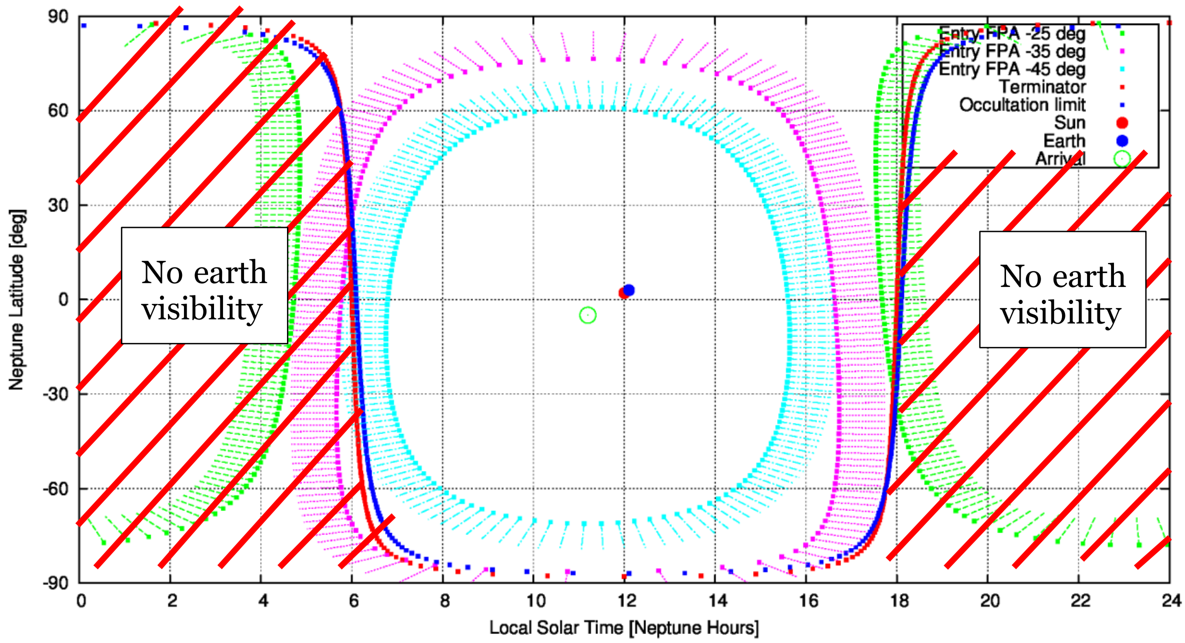


Figure 5-1: Different entry conditions depending on the flight path angle

From Figure 5-1 it is clear that the FPA of the probe should be ca. -35 deg or lower. While a -45 deg FPA can reach different latitudes of Neptune and still have Earth/ Sun visibility. However changing the -35 deg to -45 deg the FPA increases the aerodynamic flux and therefor increases the TPS. In order to reuse the PEP study heritage, the baseline for the Ice Giants probe design was selected as -35 deg FPA.

5.1.4 System Baseline Design

5.1.4.1 Mission phases

The mission phases of the probe are the following:

- Transfer phase: When the probe is attached to the orbiter. This phase lasts 13.5 years and ends when the probe is released from the orbiter.

- Coast phase:
  - After the probe is released from the orbiter the probe will coast for 20 days to Neptune. Immediately after the release of the probe the system is turned "ON" for 10 minutes to checkout all of the equipment with the exception of the instruments.
  - After this 10 minute checkout, all of the equipment are turned "OFF" with the exception of a small MTU timer which will wake up the system approx. ~30 minutes before entry.
  - Approx. 30 minutes before entry, the probe equipment and instruments will be turned "ON" for a final pre-entry checkout and to calibrate the GNC system. Note that this duration could be significantly reduced, pending a consolidated checkout timeline (thus saving a significant amount of battery energy).
- Entry phase: The probe entry phase will last ~6 min during which the parachutes are deployed and the front and back shield are released from the descent module.
- Descent phase: During the 1.5 h descent phase the probe will take scientific measurements and transmit them back to the orbiter for relay to Earth.

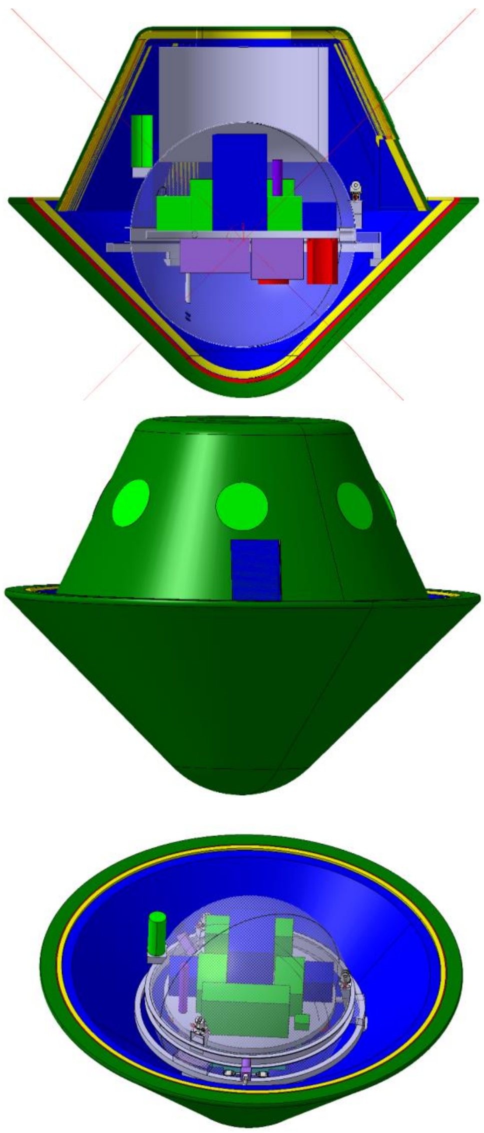
In the future, the coast phase can be optimised with respect to readout of the equipment. Currently it is assumed that the HK and instrument HK will be transmitted to the orbiter at the end of the coast phase. This does not leave any time to transmit the data back to ground and perform any error correction that might be needed. As such this data will only be used for calibration on ground.

5.1.4.2 System Modes

The system modes of the probe that were taken into account during the CDF study to model the probe are the following.

Cruise mode	•Probe carried by the Orbiter. Power interface to the orbiter for battery charging and check-ups. •*Note: Not modelled in OCDT*
Coast mode	•From probe release from orbiter until the atmospheric entry. The probe uses its own power system and timer switches to activate automatic sequences. All other units off. *Note: NO telecommand capability assumed*
Intermediate mode	•The mode in which the probe relays housekeeping data during coast phase. Used for checkout and possible calibration. This mode occurs 10 minutes immediately after release from the orbiter, and from 30 minutes before atmospheric entry up until the release of the front shield after entry (app. 6 mins).
Descent mode	•After the front shield release the Descent Module is ready to: - Perform scientific measurements - Relay data

5.1.4.3 Probe baseline design

Probe		
Mass (Incl 20% system margin)	Mass w/o TPS&TC: 191 kg TPS&TC: 151 kg Mass with TPS: 342 kg	
Ballistic coefficient	Projected area: 1.43 m^2 (diameter 1.35 m) Front shield area: 1.99 m^2 Cd: 1.07 BC: 228 kg/m^2	
Payload	• Atmospheric Structure Instrument • Camera-Radiometer • Mass Spectrometer • Photometer • USO-Doppler	
EDS	2 subsonic parachutes: pilot (M=0.8) and main	
TPS	Front shield: 51.9 mm thickness, 129 kg Back shield: 31.4 mm, 19.9 kg	
GNC	2 redundant IMUs 2 parachute deployment switches	
Mechanisms	Back and Front Shell Separation Mechanisms Parachute Swivel Mechanism Mortar parachute pyro cutter Spin Eject Mechanism [Probe side]	
Communications	UHF redundant chain Patch antenna on the backshell Helix antenna during descent	
Power	4 x 3 kg batteries PCDU	
Data Handling	CDMU including timer	
Structures	61.3 kg of structures (of which 28.5kg are in the descent module)	
Thermal	31 RHUs, MLIs, Front shield radiator, pressure vessel insulation	

5.1.5 System Budgets

5.1.5.1 Mass budget

The baseline mass budget for the probe is presented in Table 5-4. It should be noted that the EDL, TPS and Structures subsystems were designed assuming a dry mass incl. TPS of 345 kg. The mass budget for the descent module only is presented in Table 5-5.

Probe Mass Budget		**Mass [kg]**
Guidance Navigation and Control		1.68
Communications		12.31
Data-Handling		1.00
Instruments		11.10
Mechanisms		9.71
Power		21.76
Structures		61.24
Entry, Descent and Landing		18.43
Thermal Control		14.19
Harness	5%	7.87
Dry Mass w/o System Margin		**159.28**
System Margin	20%	31.86
Dry Mass incl. System Margin		**191.13**
Thermal Protection		151.25
Dry Mass incl. TPSS		**342.38**

Table 5-4: Probe mass budget

Below is the descent module mass budget. This is the part of the probe that will continue the descent after the TPS has been released.

DM Mass Budget		**Mass [kg]**
Guidance Navigation and Control		1.68
Communications		8.13
Data-Handling		1.00
Instruments		11.10
Mechanisms		3.08
Power		21.76
Structures		28.52
Thermal Control		10.77
Harness	5%	4.47
Dry Mass w/o System Margin		**90.52**
System Margin	20%	18.10
Dry Mass incl. System Margin		**108.62**

Table 5-5: Descent Module mass budget

The corresponding equipment list is presented in Table 5-6.

	#	Mass (kg)	Total Mass (kg)	Mass margin (%)	Total mass incl. margin (kg)
Probe (Probe)					
COM			**11.00**	**11.86**	**12.31**
DM (Descent Module)			**7.20**	**12.85**	**8.13**
RFDN_UHF (UHF Radio Frequency Distribution Network)	1	0.50	0.50	10	0.55
UHF_LGA_Helix (UHF Low Gain Antenna)	1	1.50	1.50	5	1.58
UHF_SSPA (UHF Solid State Power Amplifier)	2	0.80	1.60	5	1.68
UHF_TX (UHF Transmitter)	2	1.80	3.60	20	4.32
Outside Descent Module					
UHF_LGA (UHF Patch LGA)	1	3.80	3.80	10	4.18
DH			**0.83**	**20**	**1.00**
DM (Descent Module)			**0.83**	**20**	**1.00**
CDM_2 (Computer and Data Management Probe #2)	1	0.83	0.83	20	1.00
INS			**9.25**	**20**	**11.10**
DM (Descent Module)			**9.25**	**20**	**11.10**
ASI (Atmospheric Structure Instrument)	1	1.25	1.25	20	1.50
Cam_Rad (Camera-Radiometer)	1	1.20	1.20	20	1.44
Mass_Spec (Mass Spectrometer)	1	5.00	5.00	20	6.00
Phot (Photometer)	1	0.30	0.30	20	0.36
USO_Doppler (USO-Doppler)	1	1.50	1.50	20	1.80
MEC			**8.80**	**10.34**	**9.71**
DM (Descent Module)			**2.80**	**10**	**3.08**
BSSM_DM (Back Shell Separation Mechanism [DM side])	1	1.40	1.40	10	1.54
FSSM_DM (Front shield sep Mec [DM side])	1	1.40	1.40	10	1.54
Outside Descent Module					
BSSM_P (Back Shell Separation Mechanism [probe side])	1	0.90	0.90	10	0.99
FSSM_P (Front shield sep Mec [probe side])	1	2.40	2.40	10	2.64
SEM_probe (Spin Eject Mec [Probe side])	1	2.40	2.40	10	2.64
Pyro_1 (Pyro #1)	3	0.00	0.00	0	0.00
cutter (Mortar parachute pyro cutter)	1	0.30	0.30	20	0.36
PWR			**20.31**	**7.14**	**21.76**
DM (Descent Module)			**20.31**	**7.14**	**21.76**
Bat_Pr (Battery_Probe)	4	2.90	11.60	5	12.18
PCDU_Pr (Power Conditioning & Distribution Unit_Probe)	1	8.71	8.71	10	9.58
STR			**51.03**	**20**	**61.24**

	#	Mass (kg)	Total Mass (kg)	Mass margin (%)	Total mass incl. margin (kg)
DM (Descent Module)			**23.77**	**20**	**28.52**
DM_MP_1 (DM Mounting Platform #1)	1	1.03	1.03	20	1.24
DM_R (DM Mid Section Ring)	1	5.70	5.70	20	6.84
DM_Sh (DM Shell)	1	12.00	12.00	20	14.40
Parach_IF_1 (DM Main Parachute Supporting Structure #1)	3	1.68	5.04	20	6.05
Outside Descent Module					
BS_Cold (BS Cold Structure)	1	4.05	4.05	20	4.86
FS_Cold (Front Shield Cold Structure)	1	7.50	7.50	20	9.00
BS_DM_IF_Brkt_1 (BS To DM IF Bracket #1)	3	1.32	3.96	20	4.75
BS_Ribs_1 (BS Stiffening Ribs #1)	3	1.00	3.00	20	3.60
FS_IF_Brkt_1 (FS IF Bracket #1)	3	1.32	3.96	20	4.75
FSSR (Front Shield Separation Ring)	1	4.79	4.79	20	5.75
TC			**12.44**	**14.02**	**14.19**
DM (Descent Module)			**9.34**	**15.35**	**10.77**
NP_PV_Ins (NP_PressureVessel_Insulation)	1	5.00	5.00	20	6.00
P_RHU_01 (P_RHU)	31	0.04	1.24	10	1.36
P_RHU_support_01 (P_RHU_support)	31	0.10	3.10	10	3.41
Outside Descent Module					
NP_BC_MLI (NP_Backcover_MLI)	1	1.54	1.54	10	1.69
NP_FS_MLI (NP_Frontshield_MLI)	1	1.42	1.42	10	1.56
NP_FS_Rad (NP_Frontshield_Rad)	1	0.15	0.15	10	0.17
TP			**126.04**	**20**	**151.25**
Outside Descent Module					
NP_BC_Abl (NP_Backcover_Ablator)	1	4.33	4.33	20	5.20
NP_BC_HotStr (NP_Backcover_HotStr)	1	7.43	7.43	20	8.92
NP_BC_Ins (NP_Backcover_Insulation)	1	4.82	4.82	20	5.78
NP_FS_Abl (NP_Frontshield_Ablator)	1	97.05	97.05	20	116.46
NP_FS_HotStr (NP_Frontshield_HotStr)	1	7.86	7.86	20	9.43
NP_FS_Ins (NP_Frontshield_Insulation)	1	2.55	2.55	20	3.06
NP_HS_Instr (NP_Heatshield_Instruments)	1	2.00	2.00	20	2.40
EDL			**15.36**	**20**	**18.43**
Outside Descent Module					
MP (Main parachute)	1	14.38	14.38	20	17.25
PC (Pilot chute)	1	0.98	0.98	20	1.18
GNC			**1.60**	**5**	**1.68**

	#	Mass (kg)	Total Mass (kg)	Mass margin (%)	Total mass incl. margin (kg)
DM (Descent Module)			**1.60**	**5**	**1.68**
LN200S_1 (LN200S #1)	2	0.75	1.50	5	1.58
PAS_switch_1 (PAS Switch #1)	2	0.05	0.10	5	0.11

Table 5-6: Probe equipment list

The low mass of the DH subsystem has increased after the IFP. The mass and other changes of the DH subsystem have not been flown down into the system budgets or other subsystems.

5.1.5.2 Power budget

The duty cycles assumed during the study for the probe equipment are presented in Table 5-7. The main assumptions made were:

- All instruments are on for 5 minutes during the intermediate mode, off during coasting and on during the entire descent mode
- During coasting, the on-board computer (CDM) is in stand-by and the MTU timer is only consuming 5 mW of power. All other equipment are turned off.
- *Note that this the MTU power of 5 mW is significantly less than the initial 180 mW assumed for the trade-off in Section 5.1.3.1, and was based on the latest available data sheets.*
- Both the IMU and the communications subsystem were assumed to be on during the entire intermediate mode, however their duty cycles could be reduced once the mode is further characterised. However, it should also be noted that the communication subsystem was sized for a maximum range of 40000 km during the descent mode. The orbiter-probe ranges of the intermediate mode were not analysed, but are expected to be higher (see MA chapter 0). Nonetheless, the data generated would be low in this mode (only housekeeping), and so this would not be expected to be a driver.
- The pyro actuators for back and front shell release were assumed to only operate for 100 ms during the intermediate mode (see Mechanisms chapter 9).

The probe's power budget taking into account these duty cycles is presented in the Power chapter.

EQUIPMENT	P_ ON	P_ST BY	REDUNDANCY SCHEME	REDUNDANCY TYPE	REDUNDANCY .K*	REDUNDANCY. N*	P_DUTY_ CYC PDM	P_DUTY_ CYC PCM	P_DUTY_ CYC PIM
ASI	6	0	-	-	-	-	1	-1	0.14
CAM_RAD	9.6	0	-	-	-	-	1	-1	0.14
CDM	5	0.0 05	-	-	-	-	0.7	0	1
LN200S	16	0	Active (or Hot)	External	1	2	1	-1	1
MASS_SPE C	9.6	0	-	-	-	-	1	-1	0.14

EQUIPMENT	P_ ON	P_ST BY	REDUNDANCY SCHEME	REDUNDANCY TYPE	REDUNDANCY .K*	REDUNDANCY. N*	P_DUTY_ CYC PDM	P_DUTY_ CYC PCM	P_DUTY_ CYC PIM
PCDU_PR	16.5	16.5	Active (or Hot)	Internal	-	-	1	-1	1
PHOT	1.2	0	-	-	-	-	1	-1	0.14
PYRO	15	0	Passive (or Cold or Standby)	Internal	1	2	-1	-1	4.6E-6
RFDN_UHF	0	0	-	-	-	-	1	-1	1
UHF_LGA_ HELIX	0	0	-	-	-	-	1	-1	1
UHF_SSPA	266.67	0	Passive (or Cold or Standby)	External	1	2	1	-1	1
UHF_TX	5	0	Passive (or Cold or Standby)	External	1	2	1	-1	1
USO_DOP PLER	12	0	-	-	-	-	1	-1	0.14

**Redundancy k out of n: #k equipment are required to perform the mission out the #n equipment baselined*

Table 5-7: Probe equipment duty cycles

5.1.5.3 Data budget

The probe instruments data budget is presented in Table 5-8.

Probe		**Data rates per mode**	
Instruments	Data Rate (kbps)	Descent Mode	Intermediate Mode
ASI	0.16	0.16	0.16
Cam_Rad	1.75	1.75	1.75
Mass_Spec	0.13	0.13	0.13
Phot	0.00026	0.00026	0.00026
USO_Doppler	0.00	0.00	0.00
Total data rate required (kbps)		2.04	2.04
Duration (min)		90	5
Total data downloaded (Mb)		11.00	0.61

Table 5-8: Probe's instruments data budget

5.1.6 Comparison with Galileo probe and PEP

Based on a request during the study, a comparison was made between the design of the Ice Giants Neptune probe and the designs of the Galileo probe and the PEP study. The investigation sought to understand why the Galileo probe could include approximately three times as much payload mass as Ice Giants, for an equivalent total probe mass.

The main findings are summarised in Table 5-9.

		PEP (Neptune)	Galileo probe (Jupiter)	Ice Giants (Neptune)
Science	Payload mass (kg)	12.3	**30**	**11.1**
	Downloaded data (Mb)	11	**3.6**	**11**
	Descent duration (min)	60+30	**60**	**90**
	Max. pressure reached (bar)	100 (free fall from 10 bar onwards)	24	10
System	Total probe mass (kg)	313	338	342
	Descent Module (kg)	165	126	111
	Parachute (kg)	8.4	**8.2**	**18.4**
	TPS (kg)	88	169	151
	Diameter (m)	1.25	1.25	1.35
	Batteries + PCDU (kg)	22	**13.5**	**22**

Table 5-9: Comparison of Ice Giants design against PEP and Galileo

For the difference with the Galileo probe, the most significant contributing factors for the difference in payload mass available is seen to be in the power subsystem and parachute design.

The Galileo probe descent timeline foresaw a descent duration of 60 mins, during which it fell from 0.4 bars to ca. 24 bars. As such, it experienced a much more rapid descent than Ice Giants, and so Galileo could use a smaller parachute.

In addition, the longer operations time of 90 mins for Ice Giants (factor 1.5), coupled with a much higher data rate (factor 2), necessitates a much larger battery than for Galileo. Notably, Ice Giants transmits about three times as much total data during the descent period (11 Mb compared to 3.6 Mb for Galileo).

For this comparison with the Galileo probe it is also worth noting the different maturity of the mass budgets being compared. While the Galileo probe figures correspond to the flown capsule, the Ice Giants mass figures for the probe are the result of a first estimate a phase 0 level. As such, the inherently carry high margins which would then be diminished as the project evolves in maturity. Therefore, a comparison between the mass figures of a flown probe and those of the very preliminary design presented in this report can serve as a guideline but nothing more than that.

The comparison with PEP notes that the PEP study used very optimistic data for the TPS performance. As such, the TPS mass for PEP is considerably smaller. The rest of the mass differences with comparison to Ice Giants are primarily driven by the change in the probe descent profile.

5.2 Orbiter

5.2.1 Mission and System Requirements

Mission Requirements		
Req. ID	**Statement**	**Parent ID**
MIS-010	The mission shall be launched within a timeframe of 2029 to 2034.	
MIS-020	The mission shall be compatible with launch via a SLS Block 1B.	
MIS-030	The total mission cost shall be within an M-class ESA science mission budget.	
MIS-040	The mission shall be capable of performing in-situ and observational measurements at Neptune with a payload suite defined by the Study Science Team.	
MIS-050	The mission shall be capable of performing in-situ and observational measurements at Triton during flybys, with a payload suite defined by the Study Science Team.	
MIS-060	The mission shall include at least do 2 years (TBC) of science operations at Neptune.	

System Requirements		
Req. ID	**Statement**	**Parent ID**
SYS-010	The orbiter shall be compatible with a dual launch with a NASA orbiter on an SLS Block 1B in a TBD configuration.	
SYS-020	The orbiter shall be delivered to NASA for final integration onto the launcher.	
SYS-030	The orbiter delivery date to NASA shall be compatible with the selected launch date and any pre-launch activities agreed between ESA and NASA.	
SYS-040	The orbiter shall be compatible with a storage of TBD months before integration onto the launcher.	
SYS-050	The orbiter design shall allow late access for integration of the RTGs under the launcher fairing.	
SYS-060	The orbiter shall include a payload suite of 116 kg (TBC).	
SYS-070	The operational lifetime of the orbiter shall be at least 15.5 years after launch.	
SYS-080	[DELETED]	
SYS-090	The orbiter shall rely solely on its own power source(s) during cruise with the NASA orbiter.	
SYS-100	The orbiter shall provide a SpaceWire interface to the NASA orbiter.	
SYS-110	The orbiter shall be asleep during cruise with the NASA orbiter, apart from periodic checkouts.	
SYS-120	The orbiter shall not require any active thermal control from the NASA orbiter.	

System Requirements		
Req. ID	**Statement**	**Parent ID**
SYS-130	The orbiter shall separate from the NASA orbiter before the Jupiter swing-by.	
SYS-140	The orbiter shall be able to perform an independent interplanetary transfer from separation until Neptune.	
SYS-150	The orbiter shall be able to insert into orbit around Neptune.	
SYS-160	The orbiter shall be able to download all gathered science data within the nominal mission duration (TBC).	
SYS-170	The orbiter shall be compatible with all environments from integration until EOM.	
SYS-180	The orbiter shall include redundancy for all mission-critical functionalities (TBC).	

Table 5-10: System requirements

5.2.2 Design Drivers

The orbiter design was mostly driven by the far astronomical distance to Earth during the science operations phase. This necessitated the use of radioisotope power sources, and put large constraints on the available data rates for science data downlink.

In addition, the long mission lifetime and close swing-by to Jupiter (in particular for the mission to Uranus) requires High Reliability parts with significant Radiation Hardness.

5.2.3 System Assumptions and Trade-Offs

Assumptions	
1	The ESA orbiter shall be launched together with a NASA orbiter in a stacked configuration. The ESA orbiter shall be topmost on the stack.
2	The launch shall take place on an SLS Block 1B.
3	The two orbiters shall remain together (and also attached to the SEP stage) until shortly before a Jupiter swing-by in December 2032. Note that this study focuses on the timeframe after the separation of the two orbiters. The preceding mission phases are not considered in detail.
4	There is no power interface to any NASA elements during cruise.
5	There shall be a SpaceWire (data) interface to the NASA orbiter for periodic checkouts during cruise.
6	The ESA orbiter shall be asleep during cruise, apart from periodic checkouts and for the preparation of the separation.
7	All communications to Earth from the orbiter during cruise shall be transmitted via the NASA orbiter / SEP stage.
8	For communications during the pre-separation activities, telecommunications to Earth from the ESA orbiter would be either via the NASA orbiter / SEP stage, or via the ESA orbiter during dedicated reorientations of the stack.
9	Up to 3 eMMRTGs would be available from NASA for the ESA orbiter.
10	The spacecraft structure and the equipment casing can provide up to 4 mm of radiation shielding

5.2.3.1 Dual launch configuration trade-off

Two options for the dual launch configuration were preliminarily assessed for feasibility. In the first option, the ESA orbiter was considered to sit above the NASA orbiter/SEP stage stack (see Figure 5-2). In the second option, the two orbiters would be launched side-by-side atop the SEP stage (see Figure 5-3). Note that for both cases, the short Payload Fairing concept for the SLS Block-1B was used (as defined in RD[8]).

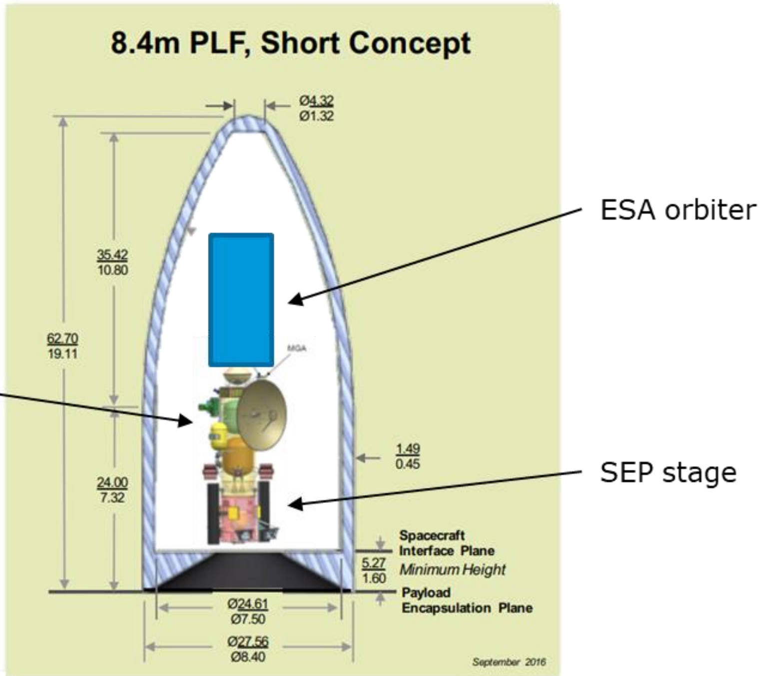


Figure 5-2: Dual launch – stacked configuration (includes images adapted from RD[8] and RD[1] for illustrative purposes)

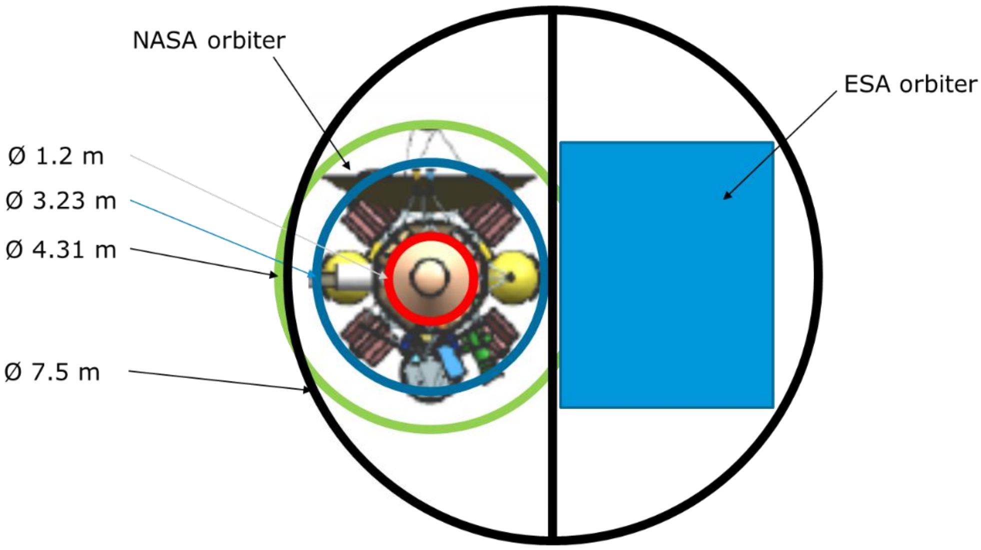


Figure 5-3: Dual launch – side-by-side configuration (includes image adapted from RD[1] for illustrative purposes)

The initial sizing of both options suggested that, from a configuration point-of-view, both alternatives would be feasible. Various benefits and risks were identified for both configurations, however a more detailed analysis (in combination with NASA) would be required to decide upon the final flight configuration. Several issues, such as the coupled mechanical loads, access under the launcher fairing for the RTGs installation, attitude control during cruise and the risk of non-separation (or from misalignments during separation) would need to be studied at much greater depth. The stacked configuration was selected as baseline for the remainder of the study.

5.2.3.2 Radiation shielding

The radiation levels observed by the spacecraft equipment behind 2.5, 4 or 10 mm of shielding structure are presented in Table 5-11. To protect the orbiter's equipment from these radiation levels, 3 options were considered:

1. Shield sensitive units individually
2. Perform delta-design and re-qualification of sensitive units to increase radiation tolerance
3. Shield the entire inner spacecraft (creating a shielded "vault")

	TIDL (krad)		
	2.5 mm (~100mil)	4 mm	10mm
Neptune	133	66	21

Table 5-11: Radiation levels for Neptune Orbiter

It was assumed for the analysis that the spacecraft structure and the individual equipment casing can provide 4 mm of radiation shielding to each unit. Table 5-12 presents an overview of the amount of extra shielding required (Option 1), or, alternatively, which units would require delta-design/re-qualification (Option 2). For Option 3, using a spacecraft diameter of 3.5m and height of 2.1m, the aluminium mass required for a full body shielding was estimated to be 374 kg (including 20% margin). As such, Option 3 was discarded.

From Table 5-12, and given the known TIDS, only 3 equipment for the Neptune orbiter were estimated to require extra shielding or modification/re-qualifications. However, the TIDS of the communication subsystem equipment and payload were not possible to identify during the course of the study. In a worst case scenario, they might require 150 kg of radiation shielding or modification/re-qualification. Thus, the TIDS of these units should be addressed in future work.

Unit	TIDS (krad)	Neptune Option 1			Neptune Option 2
		Thickness required	Thickness applied (assuming 4 mm provided by structure and unit)	Aluminium Radiation shielding mass (kg)	Delta-design/ re-qualification required to 66 krad
Gyro	20	>10 mm	6 mm	1.3	Yes
IMU	100	None	None	0	No
NavCam	2000	None	None	0	No
RW	20	>10 mm	6 mm	4x2.9	Yes
STR	2000	None	None	0	No
CDMU	100	None	None	0	No
RIUC	100	None	None	0	No
PCDU	50	>10 mm	6 mm	6.7	Yes
Batteries	4000	None	None	0	No
Radiation monitor	100	None	None	0	No
Comms*	TBD	>10 mm	6 mm	20	TBD
Payload	TBD	>10 mm	6 mm	130	TBD
Total				174	
Total w/ 20% margin				207	

*includes shielding of Ka and X-band EPC, TWT and X-band Transponder

Table 5-12: Neptune orbiter radiation trade-off

Note that for the design baseline and mass budget, it was assumed that all units would be able to reach a TIDS of minimum 60 krad. This would correspond to the 4 mm Al case, excluding RHA margin. As such, a delta-design/re-qualification of at least the Gyros, Reaction Wheels and PCDU would be required. Note that this delta-design/re-qualification should actually target at least 66 krad, if the RHA margin is to be applied.

5.2.4 Mission System Architecture

5.2.4.1 Mission timeline

The mission timeline is presented in Figure 5-4. Note that the proposed mission foresees two orbiters, one provided by ESA (which journeys to Neptune) and one provided by NASA (which journeys to Uranus).

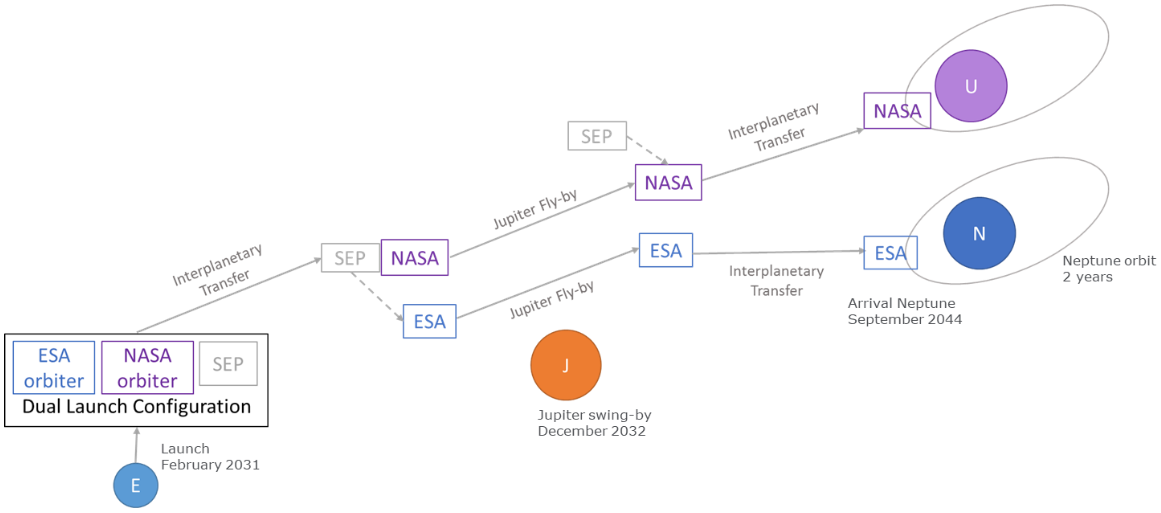


Figure 5-4: Mission timeline (Neptune)

The two orbiters are to be launched in a dual launch configuration on an SLS Block-1B in February 2031. The NASA orbiter is assumed to be attached to a Solar Electric Propulsion (SEP) stage, which provides power (to the NASA orbiter) and propulsion (to the stack) up until just before the Jupiter swing-by in December 2032, when the ESA orbiter detaches from the NASA orbiter. It thereafter travels alone to Neptune.

The ESA orbiter should arrive at Neptune in September 2044. The science operations phase at Neptune is envisaged to include planetary science of Neptune, coupled with multiple fly-bys of its moon Triton. The science phase at Neptune should last 2 years.

5.2.4.2 Mission phases

The mission phases are presented in Table 5-13. Note that the majority of the duration of the "independent swing-by phase [of Jupiter]" corresponds to the time before the Jupiter swing-by when the ESA orbiter is separated from the NASA orbiter. This phase was, however, not analysed in detail during the study.

Mission Phase	Duration
(LEOP and) transfer phase [to Jupiter]	1.5 - 2 years
Independent swing-by phase [of Jupiter]	~6 months (TBC)
Cruise phase	11.5 years
Insertion phase	1-2 weeks (TBC)
Science phase	2 years
Disposal	TBD
TOTAL:	**~15.5 years**

Table 5-13: Mission phases (Neptune)

The duration of the insertion phase (for operational constraints) and disposal are also to be clarified in later work.

5.2.4.3 Science operations timeline

The science operations to be performed in Neptune orbit include a mixture between planetary observations and measurements of Neptune's largest moon, Triton. The Triton measurements are performed during close fly-bys, while the majority of the Neptune science is also performed at Neptune periapsis. A reference science timeline was defined in order to size the system, as represented in Figure 5-5. Note however that this does not correspond precisely to any single orbit case identified by mission analysis.

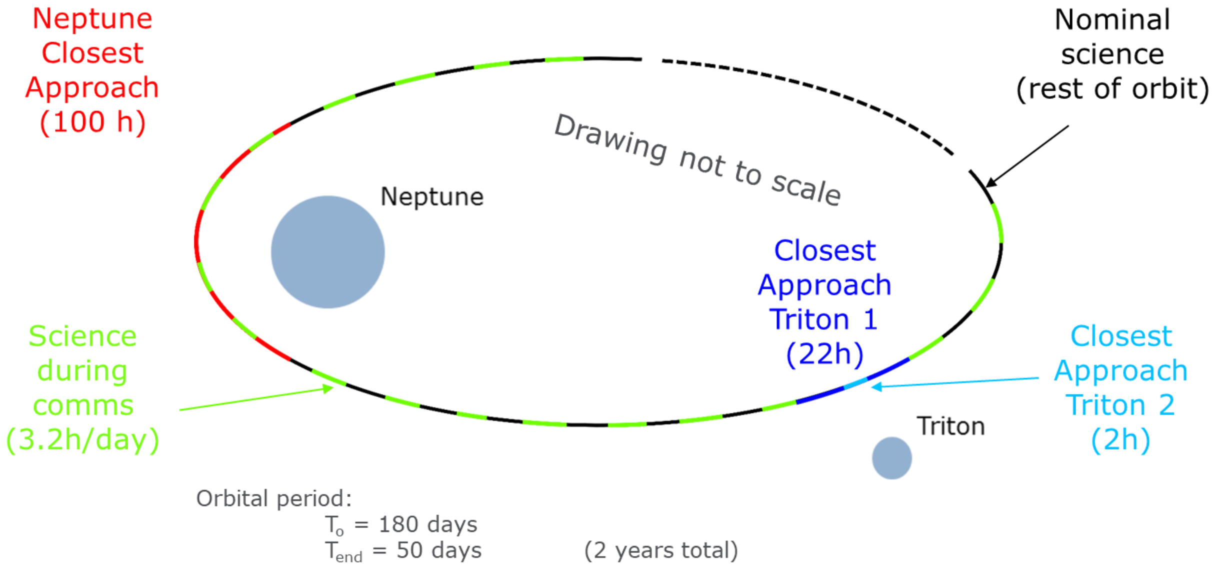


Figure 5-5: Science operations timeline (Neptune)

The reference orbit includes 100 hours of Neptune periapsis science, during the "Neptune Closest Approach" phase. This is broken only intermittently by 3.2 hour communications windows, budgeted for one window per 24 hours.

During the Triton fly-bys, there are two science phases: the "Closest Approach Triton 1" (11 hours before and after the closest point to Triton), and the "Closest Approach Triton 2" (the 2 hours closest to Triton during the fly-by). This covers a total period of 24 hours of Triton science. During this phase, there are no communications back to Earth.

For the remainder of the orbit, the orbiter performs "Nominal science". This uses a reduced payload complement, in order to use the surplus power from the RTG's to charge the spacecraft batteries for the higher-consumption phases. During this part of the orbit, the spacecraft also performs 3.2 hours communications of science data per day.

Note that the reference timeline considers a reduction in the orbital period over the 2 years of science performed at Neptune. This reduces from an orbital period of 180 Earth days at the initial orbit, to 50 Earth days by the end of the mission. This was sized on a preliminary understanding of the Triton tour envisaged by NASA in RD[1]. Later analysis revealed that this assumption was incorrect, and that an orbital period reduction of initially 35 Earth days down to 5 Earth days by end-of-mission was more likely. This however could not be addressed in the current work. Such sizings are highly dependent on the Triton fly-by tour selected, and as such this should be further iterated in future work.

5.2.4.4 System modes

The orbiter system modes are defined in Figure 5-6.

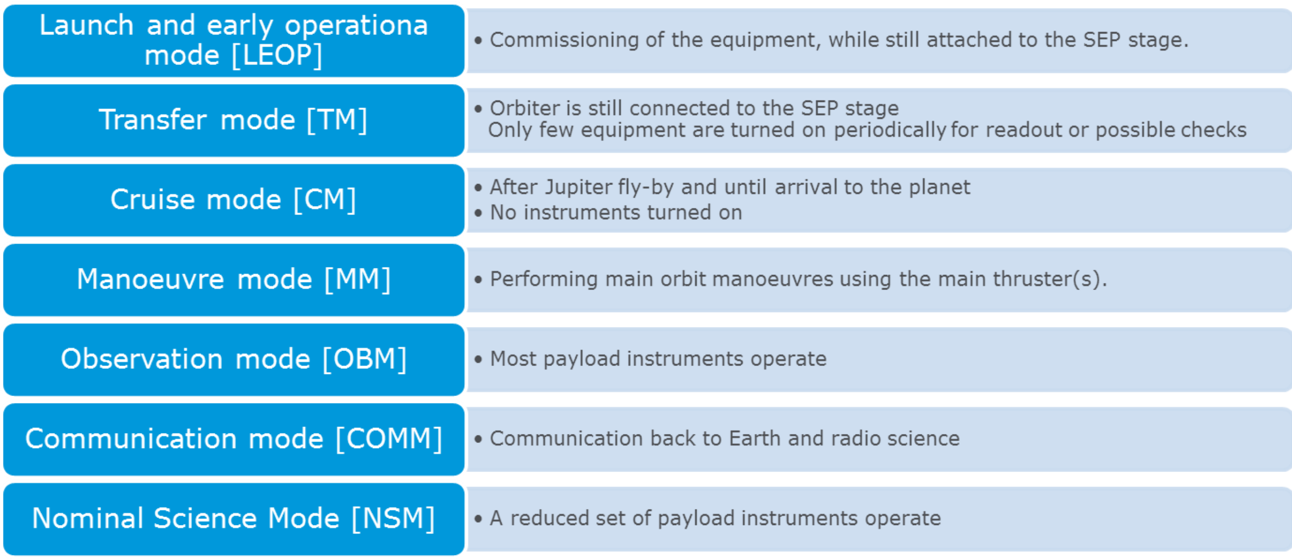


Figure 5-6: System modes (Neptune)

5.2.5 System Baseline Design

The baseline orbiter design is summarised in Table 5-14.

Orbiter	
Mass (Incl 20% system margin)	Dry mass: 1605 kg Propellant mass (excl. margin): 1991 kg Wet mass: 3969 kg
Payload	Camera Imaging Spectrometer Ion and Neutral Mass Spectrometer Magnetometer Macrowave radiometer Ultra Stable Oscillator (USO) Ka-band transponder
Propulsion	2x main bipropellant thrusters (1000 N) 16x RCS thrusters (10 N) 3x pressurant tanks (2x 120 L and 1x 66 L tanks) 4x propellant tanks (550 L)
AOGNC	1x coarse rate sensor 2x navigation cameras

Orbiter	
	2x IMUs 2x star trackers 4x reaction wheels (+ RCS thrusters)
Communications	X-band uplink/downlink Ka-band downlink (42 kbps) Science volume downlinked: 0.48 Gb/day Communication window duration: 3.2 h/day Data volume generated by EOM: 350 Gb
Power	3x eMMRTGs (EOM Power = 90W) 4x 48kg batteries
Data Handling	Redundant OBC + 1Tbit of storage
Structures	303 kg
Thermal	Heaters + use of the eMMRTG thermal dissipation

Table 5-14: Orbiter system baseline (Neptune)

5.2.5.1 Margin policy

The margin policy used in this study is the CDF margin policy for science missions. The following points note either exceptions or deviations from the standard policy.

5.2.6 System Budgets

5.2.6.1 Mass budget

The mass budget for the Neptune orbiter is presented in Table 5-15. The propellant mass is based on a total delta-v of 2712 m/s. The mass margin for the propellant residuals is already included (see Chemical Propulsion Chapter).

SC Mass Budget		Mass [kg]
Attitude, Orbit, Guidance, Navigation Control		60.40
Communications		71.64
Chemical Propulsion		224.95
Data-Handling		38.48
Instruments		118.41
Mechanisms		39.00
Power		350.04
Structures		303.26
Radiation Shielding		0.00
Thermal Control		65.89
Radiation Instrumentation		1.49
Harness	5%	63.60
Dry Mass w/o System Margin		**1337.17**
System Margin	20%	267.43

Dry Mass incl. System Margin		**1604.60**
CPROP Fuel Mass		887.53
CPROP Fuel Margin	0%	0.00
CPROP Oxidizer Mass		1464.42
CPROP Oxidizer Margin	0%	0.00
CPROP Pressurant Mass		12.03
CPROP Pressurant Margin	0%	0.00
Total Wet Mass		**3968.59**

Table 5-15: Neptune orbiter mass budget

The corresponding equipment list is presented in Table 5-16.

Equipment	#	Mass (kg)	Total Mass (kg)	Mass margin (%)	Total mass incl. margin (kg)
SC (Spacecraft)					
AOGNC			**56**	**7.86**	**60.4**
IMU_Astrix_1090A_1 (IMU Airbus Astrix 1090A #1)	2	5.00	10.00	5	10.5
NavCam_1 (NavCam #1)	2	11.00	22.00	5	23.1
RW_HR04_1 (RW Honeywell HR04 #1)	4	2.60	10.40	20	12.5
STR_HydraEU_Juice_1 (STR Sodern Hydra JUICE Electronics Unit #1)	2	3.60	7.20	5	7.6
STR_HydraOH_Juice_1 (STR Sodern Hydra JUICE Optical Head #1)	2	2.80	5.60	5	5.9
GYRO_Sireus (GYRO Selex Galileo Sireus)	1	0.80	0.80	10	0.9
COM			**64.20**	**11.59**	**71.6**
HGA (High Gain Antenna)	1	33.00	33.00	10	36.3
KaEPC (Ka-Band Electronic Power Conditioning)	2	1.30	2.60	20	3.1
KaTWT (Ka-Band Traveling Wave Tube)	2	0.80	1.60	20	1.9
LGA_LHCP (Low Gain Antenna - LHCP)	1	0.90	0.90	5	0.9
LGA_RHCP (Low Gain Antenna - RHCP)	1	0.90	0.90	5	0.9
RFDN (Radio Frequency Distribution Network)	1	13.00	13.00	20	15.6
XEPC (X-Band Electronic Power Conditioning)	2	1.30	2.60	5	2.7
XKaXPND (X/X/Ka-Band Transponder)	2	4.00	8.00	5	8.4
XTWT (X-Band Traveling Wave Tube)	2	0.80	1.60	5	1.7
DH			**32.07**	**20**	**38.5**
RIUC (Remote Inteface Unit Centralised)	1	8.33	8.33	20	10.0
RIUD (Remote Interface Unit Decentralised)	1	7.08	7.08	20	8.5
CDMU_1 (Computer and Data Management Unit #1)	2	8.33	16.66	20	20.0
INS			**98.94**	**19.68**	**118.4**

Equipment	#	Mass (kg)	Total Mass (kg)	Mass margin (%)	Total mass incl. margin (kg)
Cam (Camera)	1	16.00	16.00	20	19.2
Im_spec (Imaging Spectrometer)	1	15.50	15.50	20	18.6
INMS (Ion and Neutral Mass Spectreometer)	3	12.00	36.00	20	43.2
Mag (Magnetometer)	1	4.56	4.56	20	5.5
Micro_rad (Microwave radiometer)	1	19.34	19.34	20	23.2
USO (Ultra Stable Oscillator)	1	2.00	2.00	20	2.4
Ins_KaEPC (Instrument Ka-Band Electronic Power Conditioning)*	1	1.30	1.30	5	1.4
InsKaTWT (Instrument Ka Band Traveling Wave Tube)*	1	0.80	0.80	5	0.8
Ka_Transp (Ka-band Trransponder)	1	3.44	3.44	20	4.1
MEC			**35.00**	**11.43**	**39.0**
magBOOM (Deployable magnetometer boom)	1	30.00	30.00	10	33.0
SEP_separation (SEP stage separation [SC side])	1	5.00	5.00	20	6.0
PWR			**324.90**	**7.74**	**350.0**
Bat_Orb (Battery_Orbiter)	4	43.90	175.60	5	184.4
EMMRTG (Enhanced_Multi_Mission_RTG)	3	45.00	135.00	10	148.5
PCDU_Orb (Power Conditioning & Distribution Unit_Orbiter)	1	10.30	10.30	20	12.4
Ext_Pwr_Shnt (External power shunt)	1	1.00	1.00	20	1.2
Res_Pwr_Shnt (Resisitive power shunt)	3	1.00	3.00	20	3.6
STR			**252.72**	**20**	**303.3**
APs (Assembly Panels)	1	54.82	54.82	20	65.8
BP (Bottom Panel)	1	18.16	18.16	20	21.8
CPROP_TD (CPROP_Tank Deck)	1	21.88	21.88	20	26.3
MC (Module Collars)	1	22.00	22.00	20	26.4
SPs (Shear_Panels)	1	28.64	28.64	20	34.4
TP (Top Panel)	1	18.16	18.16	20	21.8
TR (Tube Rings)	1	12.74	12.74	20	15.3
TSS (Tank Supporting Struts)	1	66.00	66.00	20	79.2
TST (Tank Supporting Tube)	1	10.32	10.32	20	12.4
TC			**59.72**	**10.34**	**65.9**
TCS (Thermal Control Subsystem)	1		0.00	0	0.0
NO_BP (NO_Black_Paint)	1	10.00	10.00	10	11.0
NO_Louvre (NO_Louvres)	1	2.05	2.05	20	2.5
NO_MLI_ex (NO_MLI_external_22-layer)	1	32.00	32.00	10	35.2
NO_MLI_HGA (NO_MLI_HGA_10-layer)	1	1.60	1.60	10	1.8
NO_MLI_int (NO_MLI_internal_10-layer)	1	3.20	3.20	10	3.5
NO_Rad (NO_Radiator_SSM-tape)	1	0.20	0.20	10	0.2

Equipment	#	Mass (kg)	Total Mass (kg)	Mass margin (%)	Total mass incl. margin (kg)
NO_WP (NO_White_Paint)	1	0.80	0.80	10	0.9
NO_MLI_RTG_rad (NO_MLI_RTG_radiative_shield)	1	0.80	0.80	10	0.9
NO_MLI_RTG_ShuntRad (NO_RTG_ShuntRadiator)	1	1.88	1.88	10	2.1
NO_Therm_01 (NO_Thermistor)	40	0.06	2.40	10	2.6
O_Heater_01 (O_Heater)	80	0.06	4.80	10	5.3
CPROP			**213.05**	**5.59**	**224.9**
Biprop_FDV_1 (Biprop_FillDrain_Valve)	9	0.07	0.63	5	0.7
Biprop_Filter_1 (Biprop_Filter)	4	0.08	0.31	5	0.3
Biprop_LP_Trans_1 (LP_Transducer)	4	0.22	0.86	5	0.9
Biprop_LV_1 (Biprop_Latch_Valve)	4	0.75	3.00	5	3.2
Biprop_NRV_1 (Non_Return_Valve)	4	0.59	2.34	5	2.5
Biprop_Pipes (Biprop_Pipes)	1	8.00	8.00	20	9.6
Biprop_Thruster_Main_1 (Biprop_Thruster_Main #1)	2	7.80	15.60	5	16.4
Biprop_PR_1 (Biprop_PressureRegulator)	2	1.00	2.00	5	2.1
Biprop_Pres_Tank_1 (Biprop_Pressurant_Tank)	2	23.50	47.00	5	49.4
Biprop_Prop_Tank_1 (Biprop_Prop_Tank)	4	27.08	108.31	5	113.7
Biprop_SMA_Valve_1 (Biprop_SMA_Valve)	2	0.16	0.32	20	0.4
Biprop_Thruster_RCS_1_01 (Biprop_Thruster_RCS #1)	16	0.65	10.40	5	10.9
Biprop_HP_LV (Biprop_HP_Latch_Valve)	1	0.80	0.80	5	0.8
Biprop_HP_Trans (Biprop_HP_Transducer)	1	0.22	0.22	5	0.2
Biprop_Pres_Tank_small (Biprop_Pressurant_Tank_small)	1	12.00	12.00	5	12.6
Biprop_PV_1 (Biprop_Pyro_Valve)	4	0.32	1.26	5	1.3
RAD			**1.35**	**10**	**1.5**
rad_mon_ngrm (Radiation Monitor NGRM)	1	1.35	1.35	10	1.5

****These equipment are here modelled as part of the instruments (payload) as they are only required to perform radio science, but are actually integrated into the architecture of the communication subsystem***

Note: The data handling subsystem has gone through some changes after the IFP. These changes have not been flown down into the system budgets or other subsystems.

Table 5-16: Neptune orbiter equipment list

5.2.6.2 Power budget

The orbiter unit operations scheme per system mode is diagrammatically presented in Table 5-17.

System modes	LEOP	Transfer mode	Cruise mode	Maneuver mode	Observation mode	Comms mode	Nominal Science mode
Transmitter	Red	Red	Red	Red	Red	Green	Orange
Receiver	Green	Green	Green	Green	Green	Green	Green
OBC	Orange	Orange	Orange	Green	Green	Green	Orange
AOCS thrusters	Red	Red	Orange	Orange	Orange	Green	Orange
Main thruster	Red	Red	Red	Green	Red	Red	Red
PCDU	Green	Green	Green	Green	Green	Green	Green
Star trackers	Orange	Red	Orange	Green	Green	Green	Green
Reaction wheels	Red	Red	Orange	Green	Green	Green	Red
Navigation camera	Orange	Red	Orange	Green	Green	Orange	Red

Green – High duty cycle; Orange – Low duty cycle or in stand-by; Red - OFF

Table 5-17: Platform equipment operations per system mode

In addition to this, additional science sub-modes were defined to complement the orbiter system modes (see Table 5-18).

	OBSERVATION MODE			**COMMS MODE**	**NOMINAL MODE**
Science sub-modes / Instruments	**Neptune Closest Approach [IPCA] (100h*)**	**Closest Approach Triton 1 [IMCA1] (22h)**	**Closest Approach Triton 2 [IMCA2] (2h)**	**Science during comms [ISCOM] (3.2h)**	**Nominal science [IN] (remainder of orbit)**
Cam	X	O / X	O / X	O	O
Im_spce	X	O / X	O / X	O	O
INMS	O	O	O / X	O	O
KA_transp	O	O / X	O / X	X	O
Mag	X	X	X	X	X
Micro_rad	X	X	X		
USO	X	X	X	X	X

Green (X) – ON; Orange (O/X) – Low to high duty cycle; Red (O) - OFF

Table 5-18: Payload operations per science sub-mode

The duty cycles modelled in OCDT for the orbiter instruments and platform equipment are presented in Table 5-19 and Table 5-21, respectively. The redundancy scheme adopted for platform equipment is presented in Table 5-20.

The main assumptions were:

- The KA transponder, used for Doppler science during the Triton closest approach sub-modes, only operates for 1/3 of the Triton flyby, while the camera and imaging spectrometer operate for the remaining 2/3 of the fly-by.
- The instrument INMS consists of 3 units which consume in total 63.72 W. For modelling purposes only, each unit was assumed to consume 1/3 of that value.
- The propulsion latch valves were assumed to have a 2% to 5% duty cycle in several system modes, but actually only require activation once. If this is taken into account in future studies, there could be a reduction of 2.4W of power in the most driving modes.
- The RCS thrusters are assumed to have a 1% duty cycle in observation mode, communication mode and nominal science mode. According to the amount of AOCS propellant required during science operations, the RCS thrusters should only have to operate a total of 22 min in 2 years (see AOCS chapter), so a 1% duty cycle can be considered conservative if the science pointing requirements remain the same.

- During the Communication mode, Doppler science is performed with Earth, using the payload KA transponder and the communication subsystem in X and Ka band. All three links are required to perform Earth Doppler science, however not simultaneously. The X band is only required for a fraction of time (2% duty cycle), when occultations occur.
- During cruise mode, only periodic checks of the orbiter are done which require Earth pointing for data transfer. For that purpose a 10% duty cycle was assumed in that mode for the IMU, RWs and STRs.
- The heaters duty cycles assumed correspond to the required total consumption stated in the Thermal Chapter, for an amount of 80 heaters.
- The radiation monitor should be on during the transfer and science operations, but could be turned off if power is required for other activities.
- The orbiter's transmitters should be off during the LEOP mode to not blind any communications with the NASA orbiter and SEP stage, and to respect ITU requirements (this was not injected into the OCDT model, but does not affect the design).

The orbiter's power budget taking into account these duty cycles is presented in the Power chapter. It should be noted that the RCS thrusters and the second main thruster were added to the model later and thus the power budget used for designing the power subsystem does not include these thrusters (approximately 300W, including 20% system margin, are missing in the manoeuvre mode). This should however not affect the power subsystem design, which is driven by the science operations.

INSTRUMENT	P_ON	P_STBY	P_DUTY_CYC IPCA	P_DUTY_CYC IMCA1	P_DUTY_CYC IMCA2	P_DUTY_CYC ISCOM	P_DUTY_CYC IN
CAM	34.8	0	1	0.66	0.66	-1	-1
IM_SPEC	25.2	18.9	1	0.66	0.66	-1	-1
INMS	21.24	10	-1	-1	0.66	-1	-1
INMS_2	21.24	10	-1	-1	0.66	-1	-1
INMS_3	21.24	10	-1	-1	0.66	-1	-1
INS_KAEPC	3	0	-1	0.33	0.33	1	-1
INSKATWT	60	0	-1	0.33	0.33	1	-1
KA_TRANSP	39.84	0	-1	0.33	0.33	1	-1
MAG	12	2.74	1	1	1	1	1
MICRO_RAD	67.14	14.04	1	1	1	-1	-1
USO	6	0	1	1	1	1	1

Table 5-19: Orbiter instruments duty cycles

EQUIPMENT	P_ON	P_STBY	REDUNDANCY. SCHEME	REDUNDANCY .TYPE	REDUNDANCY .K*	REDUNDANCY .N*
BIPROP_HP_TRANS	0.3	0.3	None	-	-	-

EQUIPMENT	P_ON	P_STBY	REDUNDANCY. SCHEME	REDUNDANCY .TYPE	REDUNDANCY .K*	REDUNDANCY .N*
BIPROP_LP_TRANS	0.8	0	Active (or Hot)	Internal	1	2
BIPROP_LV	30	0	Passive (or Cold or Standby)	Internal	2	4
BIPROP_THRUSTER_MAIN	180	0	Passive (or Cold or Standby)	External	1	1
BIPROP_THRUSTER_RCS	16.8	0	Passive (or Cold or Standby)	External	8	16
CDMU	35	0	Passive (or Cold or Standby)	External	1	2
EPC	9.07	0	Passive (or Cold or Standby)	External	1	2
IMU_ASTRIX_1090A	21	0	Passive (or Cold or Standby)	External	1	2
KATWT	172.41	0	Passive (or Cold or Standby)	External	1	2
NAVCAM	5.25	0	Passive (or Cold or Standby)	External	1	2
O_HEATER	0.6	0	-	-	-	-
PCDU_ORB	24	24	Active (or Hot)	Internal	-	-
PYRO_BOOM	15	0	Passive (or Cold or Standby)	Internal	1	2
RAD_MON_NGRM	2.65	0	None	-	-	-
RIUC	16	0	-	-	-	-
RIUD	12	0	-	-	-	-
RW_HR04	9.6	0	Passive (or Cold or Standby)	External	3	4
STR_HYDRAEU_JUICE	11.55	0	Passive (or Cold or Standby)	External	1	2
STR_HYDRAOH_JUICE	7.88	0	Passive (or Cold or Standby)	External	1	2
XEPC	5.99	0	as above	External	1	2
XPND_RX	15	0	as above	External	1	2
XPND_TX	20	0	as above)	External	1	2
XTWT	112.07	0	as above	External	1	2

**Redundancy k out of n: #k equipment are required to perform the mission out the #n equipment baselined*

Table 5-20: Orbiter equipment redundancy scheme

EQUIPMENT	P_DUTY_ CYC LEOP	P_DUTY_ CYC TM	P_DUTY_ CYC CM	P_DUTY_ CYC MM	P_DUTY_ CYC OBM	P_DUTY_ CYC COMM	P_DUTY_ CYC NSM
BIPROP_HP_TRANS	1	1	1	1	1	1	1
BIPROP_LP_TRANS	1	1	1	1	1	1	1
BIPROP_LV	-1	-1	0.02	0.05	0.02	0.02	0.02
BIPROP_THRUSTER_ MAIN	-1	-1	-1	1	-1	-1	-1
BIPROP_THRUSTER_ RCS	-1	-1	0.01	0.5	0.01	0.01	0.01
CDMU	1	1	0.6	1	1	1	0.4
EPC	-1	-1	-1	-1	-1	0.98	-1
IMU_ASTRIX_1090A	1	-1	0.1	1	1	-1	0.1
KATWT	-1	-1	-1	-1	-1	0.98	-1
NAVCAM	0.1	-1	-1	-1	0.1	-1	-1
O_HEATER	-1	0.792	0.938	-1	-1	-1	0.417
PCDU_ORB	1	1	1	1	1	1	1
PYRO_BOOM	-1	-1	-1	-1	-1	-1	-1
RAD_MON_NGRM	0.1	1	1	-1	1	1	1
RFDN	1	-1	0.1	0	-1	1	0.1
RIUC	1	1	0.6	1	1	1	0.4
RIUD	1	1	0.6	1	1	1	0.4
RW_HR04	-1	-1	0.1	1	1	1	-1
STR_HYDRAEU_JUICE	1	-1	0.1	1	1	1	1
STR_HYDRAOH_JUICE	1	-1	0.1	1	1	1	1
XEPC	1	-1	0.1	0	0	0.02	0.1
XPND_RX	1	1	1	1	1	1	1
XPND_TX	1	-1	0.1	0	-1	0.02	0.1
XTWT	1	-1	0.1	0	-1	0.02	0.1

Table 5-21: Orbiter equipment duty cycles

5.2.6.3 Data budget

The obiter's payload data budget is presented in Table 5-22. The data rates and compression rates presented on the left are the rates initially provided by the payload experts. However, the total data generated in each mode assuming these values could not be downloaded to ground, given the platform power constraints (see 5.2.7.1). As such, it has been agreed with the science team that, for the course of the study, it would be assumed that the payload would only need to download:

- 13 Gb of data from each Neptune closest approach (IPCA mode). The assumed rate was computed as follows: 10Gb/66.7h=0.15 Gb/h
- 4 Gb of data from each Triton closest approach sub-mode (IMCA1&2 modes).

Orbiter				Duty Cycle					Data rates per mode				
Instrument	Data Rate (kbps)	Compression	Data Rate after compression (kbps)	IPCA	IMCA 1	IMCA 2	ISCom	IN	IPCA	IMCA 1	IMCA 2	ISCom	IN
Radiation monitor	0.06	1	0.06	1	1	1	1	1	0.06	0.06	0.06	0.06	0.06
Camera	550	3	183	1	0.67	0.67			183	122	122	0	0
Im_spce	2870	3	957	1	0.67	0.67			957	638	638	0	0
INMS	0.43	1	0.43			0.67			0	0	0.29	0	0
INMS	0.43	1	0.43			0.67			0	0	0.29	0	0
INMS	0.43	1	0.43			0.67			0	0	0.29	0	0
KA_transp	0	1	0.00		0.33	0.33	1		0	0	0	0	0
Mag	2.41	2	1.21	1	1	1	1	1	1.21	1.21	1.21	1.21	1.21
Micro_rad	5.23	1	5.23	1	1	1			5.23	5.23	5.23	0	0
USO	0	1	0	1	1	1	1	1	0	0	0	0	0
Total data rate per mode	**(kbps)**								**1146**	**766**	**767**	**1.27**	**1.27**
	(Gb/h)								**4.13**	**2.76**	**2.76**	**0.0046**	**0.0046**
Duration (hours)									**86.9**	**22.0**	**2.00**	**3.28**	**20.7**
Total data generated (Gb)									**359**	**60.7**	**5.52**	**0.01**	**0.09**
Used for the study (Gb)									**13**	**4**			

Table 5-22: Orbiter's payload data budget

5.2.6.4 Dissipation budget

The dissipation budget for the Neptune orbiter is presented in Table 5-23, where the platform and power consumptions already include a 20% margin. The RF outputs assumed were as follows:

- Payload KaT: 35 W
- Communication subsystem KA-band: 100W
- Communication subsystem X-band: 65W.

It should be noted that the actual Safe mode (SM) consumed power and heater power were not modelled in OCDT and the value indicated in red is only an estimation based on the spacecraft Nominal Science mode (NSM).

There is also a discrepancy between the numbers here provided and those used to size the thermal subsystem (see Thermal chapter), due to the late addition of the RCS thrusters and a second main engine into the OCDT model, which increased the power consumed and dissipation. This increase is mostly in the manoeuvre mode (by approximately 300W, including 20% margin). However, this should mostly likely not impact the design of the thermal subsystem significantly, apart from the thermal aspects of the two main engines close to each other. Most heat during this mode is radiated to the outside of the spacecraft. Still, a reassessment of the radiator size should be made in future phases.

System Mode	CM	ObM	ObM	NSM	ObM	Com M	LEOP	MM	SM	TM
Science Sub-Mode		IMCA 1	IMCA 2	IN	IPCA	ISCo m				
Platform power consumption	183	218	218	155	218	409	342	730	183	175
Payload power consumption	0	199	262	22	174	145	0	0	0	0
Total consumption (W)	183	417	480	177	392	554	342	730	183	175
Instrument KaT duty cycle		0.33	0.33	-1	-1	1				
Instrument RF output		-11.7	-11.7			-35				
Comms X duty cycle	0.1	-1	-1	-1	-1	0.02	1	0	0.1	-1
Comms Ka duty cycle	-1	-1	-1	-1	-1	0.98	-1	-1	-1	-1
Comms RF output	-6.5	0	0	0	0	-99.3	-65	0	-6.5	0
Heater Power	-45	0	0	-20	0	0	0	0	-48	-38
Total output (W)	-51.5	-11.7	-11.7	-20	0	-134	-65	0	-54.5	-38
Dissipation (W)	131	405	468	157	392	420	277	730	129	137

Table 5-23: Neptune orbiter dissipation budget

5.2.6.5 Delta-v budget

The Neptune orbiter delta-v budget is presented in Table 5-24.

Delta-v Budget	Manoeuvre type	Orbiter to Neptune	Unit	Comment
Jupiter fly-by Targeting	stochastic	15	m/s	
Orbital Insertion	deterministic	2058.7	m/s	From propulsion, considering baseline T/M ratio
Triton/Uranus Moon Targeting	deterministic	226	m/s	
Planet Tour Deterministic	deterministic	65	m/s	
Planet Tour Stochastic	stochastic	20	m/s	
Planet Tour Future Design	deterministic	30	m/s	
Margin on stochastic delta-v		0	%	3-sigma values, no margin applied
Margin on deterministic delta-v		5	%	
Total det. and stoch. Manoeuvres		2533.65	m/s	
Disposal manoeuvre		10	m/s	

Delta-v Budget	Manoeuvre type	Orbiter to Neptune	Unit	Comment
Margin on disposal manoeuvre		0	%	
Total disposal manoeuvre		10	m/s	
AOCS delta-v		168.18	m/s	Margin on total propellant estimated by propulsion
Margin on AOCS delta-v		0	%	
Total AOCS delta-v		168.18	m/s	
Total delta-v w/o margin		**2592.8**	**m/s**	
Total delta-v with margin		**2711.8**	**m/s**	

Table 5-24: Neptune orbiter delta-v budget

5.2.7 System Options

5.2.7.1 Payload timeline

The communication window duration drives both the total data that can be downlinked and the total energy budget per orbit. However, the total data that can be downlinked is also a driver for the instrument design and the duration of the science modes, which in turn sizes the communication window duration. To estimate the total data that can be generated and downlinked an analysis has been made at system level using the following assumptions:

- 3 RTGs with an EoL power of 90 W each
- The duration of all science modes with the exception of the Nominal Science mode are fixed for each orbit
- A system margin of 20% is added to the total power consumed
- An efficiency of 90% is assumed for the losses inside the spacecraft, including the battery charging and discharging losses, PCU losses, harness losses, etc.
  Note: This efficiency, for ease of calculations, was assumed at the power generation side. This is a worst case that includes several factors that might not happen at the same time.
- The power of each mode is shown in Table 5-25
- The total duration of the each of the science modes is:
  - Neptune closest approach: 100 h including 4 communication windows where the science mode will switch to the science comms mode. (See Figure 5-5)
  - Triton closest approach 1: 22 hours
  - Triton closest approach 2: 2 hours
  - Nominal science mode: the remainder of the orbit with 1 comms mode/day
- The total science data generated in each science mode is:
  - Neptune closest approach: 10Gb/66hours (0.15 Gb/hour) has been requested by the project scientists. Since the duration of this mode is dependent on the

communication window duration, this number will be increased or decreased depending on the actual duration of the mode.
    - Triton closest approach 1: The triton closest approach 1 and 2 combined generate 4 Gb of data. (For ease of analysis this is assumed to be all generated during the Triton closest approach 2 mode.
    - Science comms mode: 1.265 kbps
    - Nominal science mode: 1.265 kbps
- The HK data during the orbit is 200 Mb/day
- The downlink datarate is 42kbps.

Power	Closest approach Neptune	Closest approach Triton1	Closest approach Triton2	Comms duration/day	Nominal Science	
Power instrument	145	166	218	121	18	W
Power platform	177	177	177	336	125	W
Total Power	**322**	**342**	**395**	**457**	**143**	**W**

Note: The following results change significantly with only minor changes in the power budget.

Table 5-25: Power consumed in each science mode for the instruments and the platform

5.2.7.1.1 Results

The results for the communication window duration have been sized for the 50 day orbit, to ensure that the total generated energy equals the total consumed energy per orbit.

The total communication window duration in this case is 3.2 hours. Table 5-26 shows that the worst case is the 50 day orbit and that there is energy available in case of a different orbit duration. Table 5-27 shows a negative data margin for both the 50 and 75 day orbit.

Figure 5-7 shows different data points taken for the data margin and energy margin. This shows that the data downlinked can be increased to be more than the data generated in the 75 day orbit by reducing the energy margin and by increasing the communication window.

Since it is not possible to increase the communication window in the 50 day orbit, the remaining data (7.2 Gb) will have to be downlinked after the 50 day orbit.

- All the remaining data can be downlinked in 15 days, assuming the 3.2 hours/day communication window, after the 50 day orbit if no extra data is generated (HK or science)
- If downlink HK data is generated, all the remaining science data can be downlinked in 24 days assuming the 3.2 hours/day communication window
- If both HK and science data is generated at the same duty cycle as the nominal case:
    - 3.2 hours of communication
    - 20.8 hours of nominal science

- 50 days of downlink are needed.

Energy				
Days	Generated	Consumed	Margin	
50	291600	291600	0	Wh
75	437400	425107	12293	Wh
100	583200	558614	24586	Wh

Table 5-26: The total power generated, consumed and the power margin for a 50, 75 and 100 day orbit

Data				
Days	Generated	downlinked	Margin	
50	32	25	-7	Gb
75	40	37	-3	Gb
100	48	50	2	Gb

Table 5-27: The total data generate, downlink availability and margin for a 50, 75 and 100 day orbit

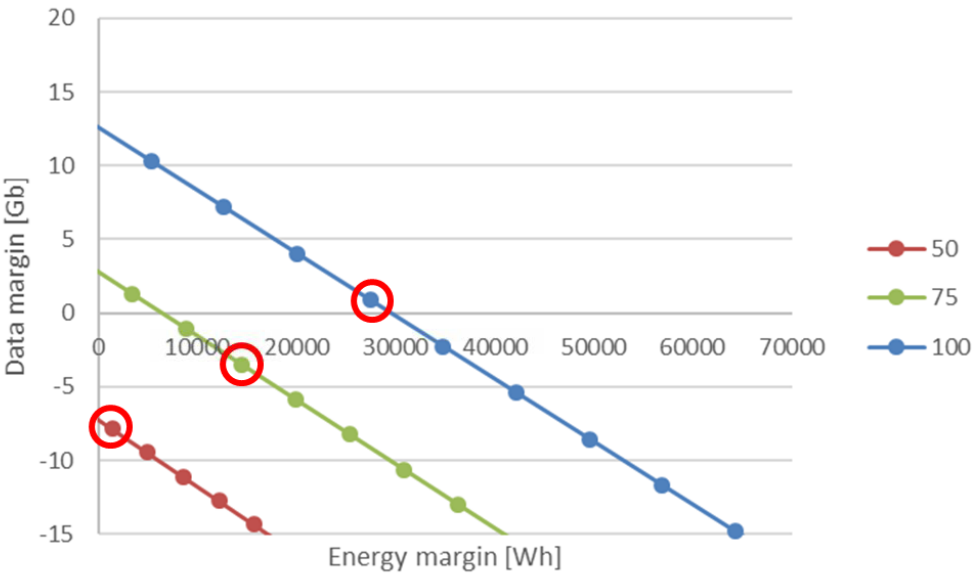


Figure 5-7: The energy margin available plotted against the data margin available. The circled points are the data points from Table 5-26 and Table 5-27

The next figures show the total power and data generated in the 50, 75 and 100 day orbit for each mode. These figures show that the driving case for the energy consumption is the nominal science mode and the communication mode. The numbers in these pictures are preliminary numbers that will change significantly with minor changes in the power budget.

For the data generated the driving case is the Neptune closest approach mode.

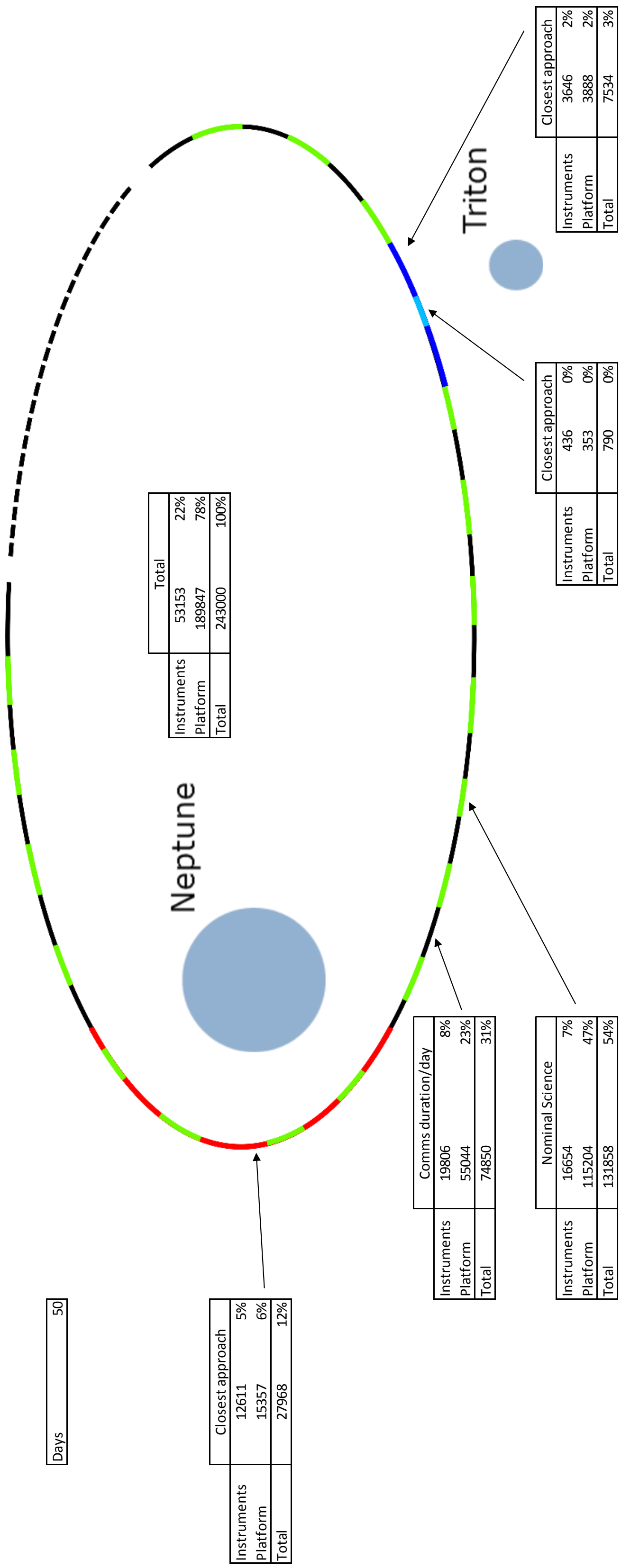


Figure 5-8: The consumed energy per mode and per orbit in kW and percentage over the total orbit for the 50 day orbit

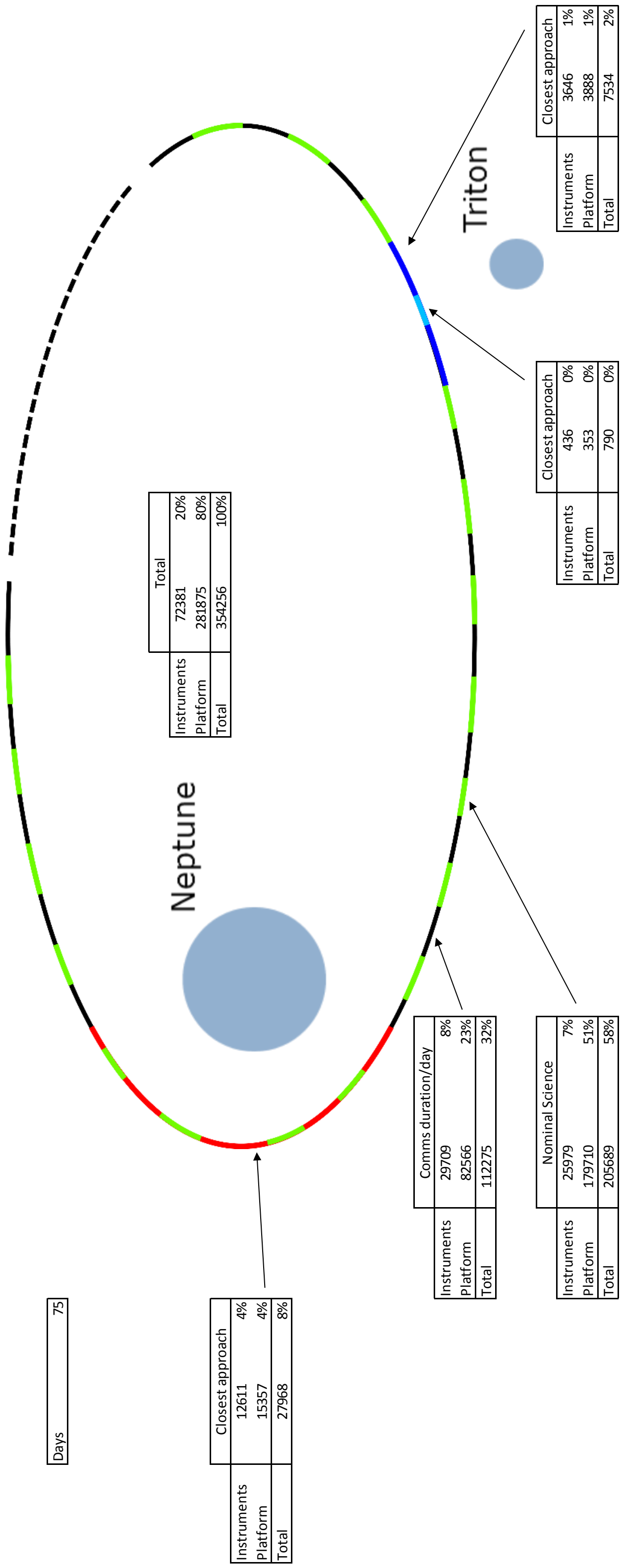


Figure 5-9: The consumed energy per mode and per orbit in kW and percentage over the total orbit for the 75 day orbit

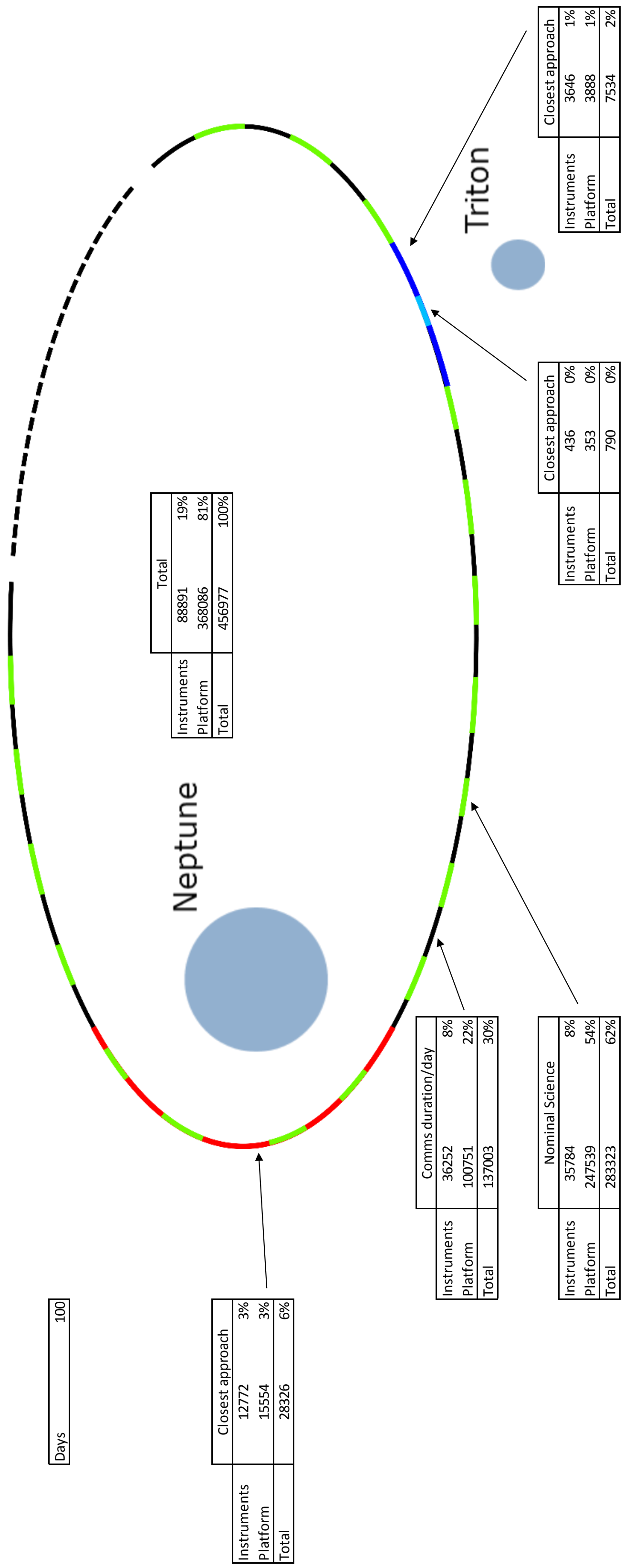


Figure 5-10: The consumed energy per mode and per orbit in kW and percentage over the total orbit for the 100 day orbit

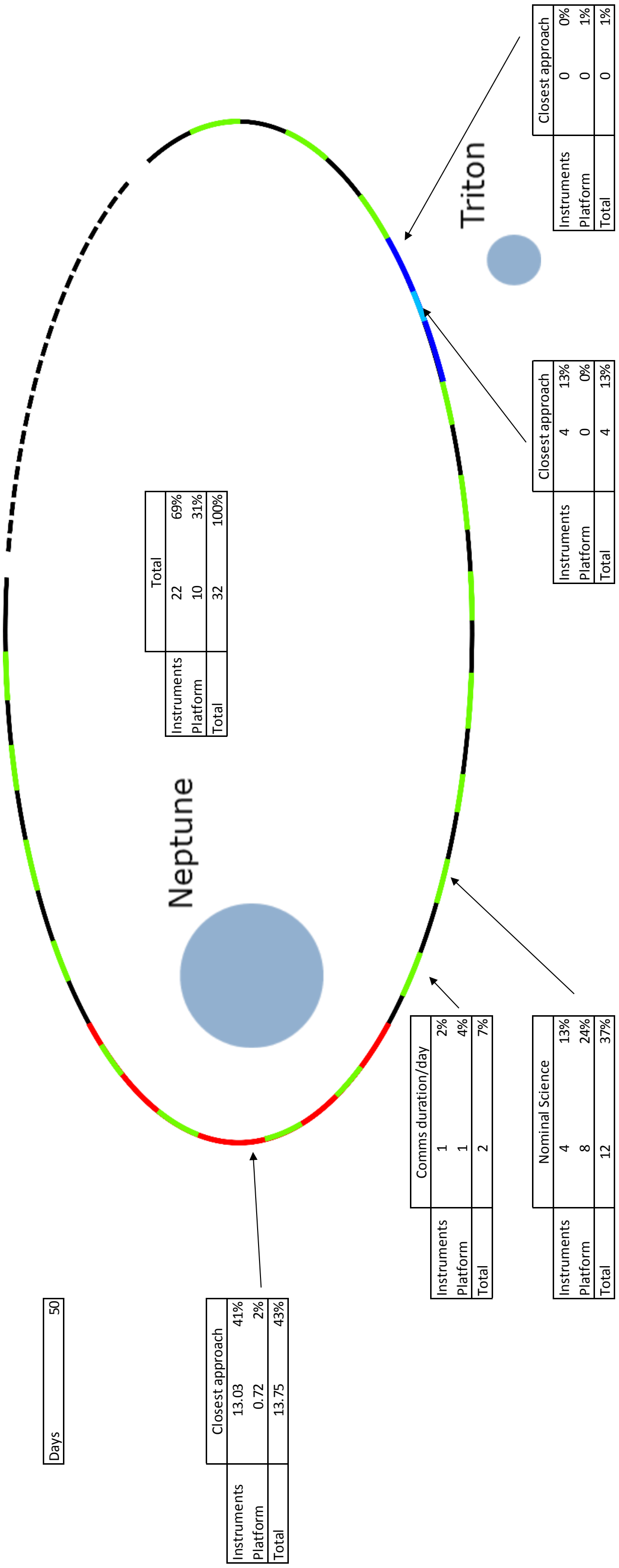


Figure 5-11: The generated data per mode in Gb and percentage over the total orbit for the 50 day orbit

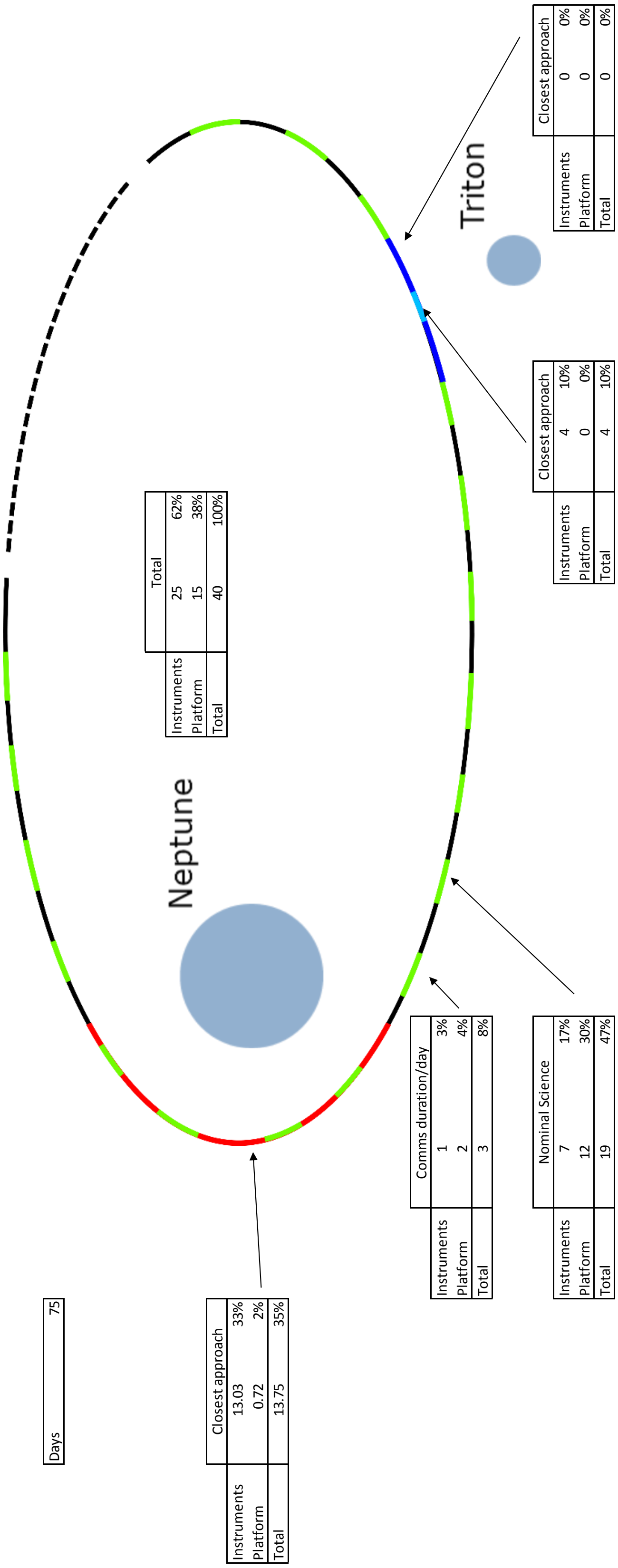


Figure 5-12: The generated data per mode in Gb and percentage over the total orbit for the 75 day orbit

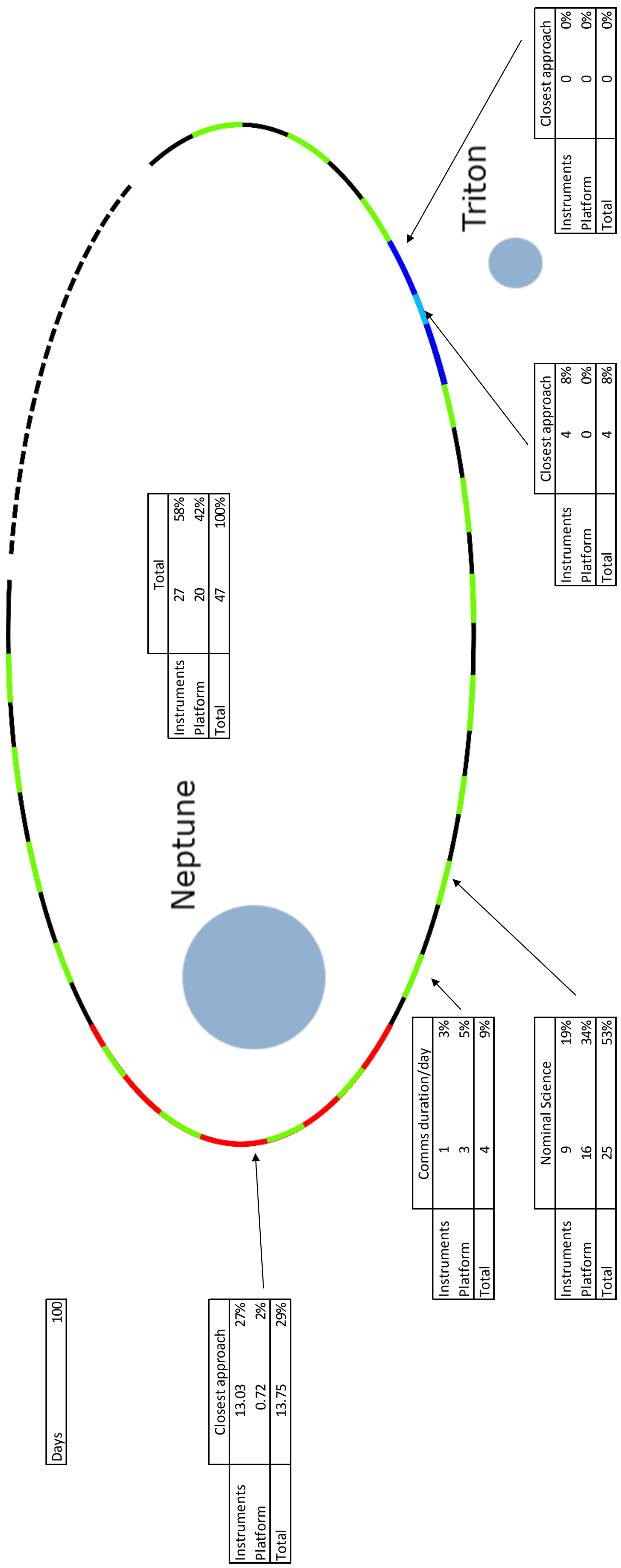


Figure 5-13: The generated data per mode in Gb and percentage over the total orbit for the 100 day orbit

5.2.7.2 Number of RTGs

In the previous analysis, the use of 3 RTGs is assumed. Since these are LLIs with considerable availability and usage challenges, the use of 2 RTGs was investigated.

Due to the already very low / negative margins on the data downlink budget it is clear that a 2 RTG solution would not be feasible for the current payload and science operations baseline.

5.2.8 Future Work

There are a number of open issues/options to be addressed in future work at system level. These trade-offs would seek to optimise the design or to mitigate identified risks and uncertainties. These include:

- **Optimisation of science timeline:**

  The Neptune mission is highly constrained by the trade-off between data downlink and power/energy. The reference case science timeline was considered to be at the margins of feasibility. As such, a detailed analysis of the desired science operations would help to reintroduce margin into the design. For instance, it could be considered to perform the majority of Neptune periapsis science during the initial Neptune orbits, which have more time to recharge the battery. The shorter Neptune orbits could then be used to focus on Triton science. Such shorter orbits were already identified in RD[1] as being of more value for Triton science.

  In addition, a simple extension of the mission duration, or intermittent breaks in high-volume science, would allow time and energy to download all acquired payload data.

- **Increasing the number or size of considered ground stations:**

  The current design considers an ESA-only array of two visible ground stations. The extension to a third ESA station in the array would offer considerable data downlink advantages, however the availability of this feature by 2044 could not be guaranteed. In addition, potential access to the 70-metre antennae of the NASA Deep Space Network could vastly increase the data throughput to Earth.

- **Consolidated analyses for launch and initial interplanetary trajectories:**

  As presented in the assumptions, the study focused on the orbiter design from the point of release from the NASA orbiter (pre Jupiter swing-by). As such, further iteration and interaction would be required with NASA to consolidate interface requirements during launch and the pre-separation cruise.

In addition, there remain significant uncertainties for the later system work:

- **Availability of Enhanced Multi-Mission Radioisotope Thermoelectric Generators (eMMRTGs):**

  The availability of 3 eMMRTGs for the ESA orbiter is critical for the mission. As discussed elsewhere in the report, the availability of these devices is combined with significant programmatic/schedule risk. There are also issues regarding access during testing and under the launcher fairing, as well as nuclear safety regulations dictating the maximum mass of radioactive material that can be stored at the launch site and launched in a single rocket (the total of 6 eMMRTGs required for this mission (3 for the ESA orbiter and 3 for the NASA one) seems to be above the allowed limit, but the indication for the purpose of this study was to not go into those details at this stage) . None of these issues were addressed in detail in this study. However, any one of them could potentially be a showstopper for the realisation of the mission. The end-of-mission output power to be expected from the eMMRTGs remains an additional point that could potentially restrain the mission science operations further.

5.2.9 Technology Needs

All technology needs are considered at subsystem level. Note however that the baseline launcher (SLS Block 1B) will not be available until after 2021 (RD[8]).

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6 NEPTUNE PAYLOAD

6.1 Atmospheric Probe

The Model Payload for the Neptune Atmospheric Probe was put together in order to size the capsule resources and to address the payload accommodation. These instruments are a representation of a possible future payload, but are not meant to be understood as a pre-selection of instruments for a potential future mission.

This Model Payload was taken over from an earlier CDF study, the *Planetary Entry Probe to Venus* ("PEP (V)"; RD[7]). This approach was chosen due to the limited study time available of five weeks for three potential mission elements (Atmospheric Probe, Orbiter, and Triton Lander) and the need to devote more time to study the mission enabling elements of the Atmospheric Probe: the critical technologies of the Entry Descent System (EDS), in particular the heat-shield and the parachute(s).

6.1.1 Requirements and Design Drivers

The main design drivers for the Model Payload of the Atmospheric Probe were the following:

- The instruments shall survive a ~13-year transfer to Neptune. Throughout the transfer the temperature of the instruments shall be sufficient (value TBC) to keep them functioning optimally for operations at the target planet.
- The instruments shall have sufficient power for science operations during the 90-minute descent of the Atmospheric Probe down to 10 bar (minimum) in Neptune's atmosphere (see *Power section 13.1.3*
- The instruments shall be able to uplink the science data in real time to the communications system of the relay satellite.
- Instruments that will start operations before the start of real-time uplink shall be able to store the acquired data in a data storage unit and this data shall be transmitted by the communications system at the appropriate time.

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
PAY-010	The Atmospheric Probe instruments shall be able to operate between 0.1 bar and 10 bar (minimum).	TBD
PAY-202	The Atmospheric Probe instruments shall be able to operate between 60 and 90 minutes in Neptune's atmosphere.	TBD

Table 6-1: SubSystem requirements for the Neptune Probe Model Payload

6.1.2 Assumptions and Trade-Offs

For the Atmospheric Probe Model Payload no trade-offs were performed, as the same instruments were used in RD[7].

The assumptions that were taken are listed in Table 6-2:

Assumptions	
1	No radiation shielding is assumed as the calculated TID (Total Ionising Dose) for the Neptune mission is ~66 krad with 4 mm Al shielding see Chapter 35 Radiation.
2	Heating for the instruments shall be provided by RHUs [35.2.2].

Table 6-2: Assumptions for the Neptune Atmospheric Probe Model Payload

6.1.3 Baseline Design

The Model Payload for the Atmospheric Probe addresses science goals as described in the Science Traceability Matrix (STM). The main science objectives at Neptune are to:

- Determine the compositional, thermal and dynamical structure of the atmosphere
- Determine the planet's bulk composition, including abundances and isotopes of heavy elements.

The main instruments to address the planet's bulk composition are the Mass Spectrometer (measuring the atmospheric composition) and the Atmospheric Structure Instrument, providing supporting information on altitude profile (e.g. by pressure) and on the thermal condition, allowing for derive mixing ratio profile and detect possible condensation.

The structure of the atmosphere will be addressed by the Atmospheric Structure Instrument, Camera/Radiometer, Photometer, and the USO/Doppler wind experiment.

Table 6-3 lists those instruments, together with their mass (incl. 20% equipment margin), average power consumption, data rate, physical size and their heritage from previous instruments and missions.

Instrument	Mass [kg]	Power [W]	Data rate [kb/s]	Volume envelope [mm]	Notes/Heritage
Atmospheric Structure Instrument (ASI)	1.50	6.00	0.16	TEM, PPI: 205×30 Ø ACC: 79×58×68	Three core sensor packages: - three-axial accelerometer (ASI-ACC) - pressure profile instrument (ASI-PPI) - temperature sensors (ASI-TEM)
Mass Spectrometer	6.00	9.60	0.13	200×200×100	Ion Trap Mass Spectrometer, Rosetta/Ptolemy heritage
Ultra Stable Oscillator (USO)/Doppler wind experiment	1.80	12.00	n/a	150×150×118	USO for Doppler Wind Experiment; Huygens heritage
Camera/radiometer channels	1.44	9.60	1.747	100×100×200	For atmospheric cloud features; 17.5° FoV, 4 filters; VenusExpress heritage
Photometer	0.36	1.20	0.00026	30×30×80	Selected as placeholder for potential other instruments (see Section 6.1.3.1).

Table 6-3: Baseline Model Payload for the Neptune Atmospheric Probe

The total mass of this Model Payload amounts to 11.1kg, including 20% maturity margin with an average power consumption of ~42W (no margin included).

6.1.3.1 Payload Components

More details on some of the instruments are listed here below:

- **ASI**: The three-axial accelerometer (ASI-ACC) could possibly be replaced by a system inertial measurement unit, part of the on-board GNC (Guidance & Navigation Control) system.
- **Mass Spectrometer**: The instrument could be equipped with a gas chromatograph and a tuneable laser for high accuracy determination of noble gas and isotopic abundance/ratios.
- **Camera**: A calibrated imager could be used to study atmospheric properties, e.g. optical depth, distribution and properties of aerosols and clouds particles. Radiometer channels and possibly also V-IR spectral channels, as per Huygens DISR, could be added in order to measure thermal up- and down-flux, and atmospheric composition, respectively.
- **Photometer:** The photometer here acts as a resource placeholder for e.g. a sun-sensor to study the atmospheric optical depth and gather information on the distribution and properties of aerosols and clouds particles. A radiometer for measuring the up- and down-flux could also be used to investigate the radiative energy and thermal balance of the atmosphere. The photometer could also be replaced by a Nephelometer to sound the cloud structure and solid/liquid particles.

6.1.4 List of Equipment

See Table 6-3.

6.1.5 Options

As mentioned before, this CDF study used a Model Payload defined for a Venus Entry Probe study (RD[7], see Section 6.1), due to limited study time available. However, a dedicated Model Payload was also derived from the Science Traceability Matrix (Atmospheric probe STM,), as defined by the Study Science Team (SST). This payload is similar to the one that has been proposed by an international team of experts to explore in situ the atmospheres of Saturn and the Ice Giants (RD[9], RD[10], RD[11]).

This augmented Model Payload comprises additionally a Helium abundance detector. This augmented Model Payload comprises additionally a Helium abundance detector to detect this element in the atmosphere, as well as a Nephelometer to investigate cloud locations and aerosol properties. The Mass Spectrometer (MS) of this payload is more powerful, but also has a higher mass than the MS in Table 6-3. The Camera was replaced by a Net-flux Radiometer.

The instruments of the augmented Model Payload are listed in Table 6-4:

Instrument	Mass [kg]	Power [W]	Data rate [kb/s]	Volume envelope [mm]	Notes/Heritage
Mass spectrometer	18.96	81.60	2.00	245×145×229	Time-of-flight mass spectrometer with varying measurement cadence, tuneable laser spectrometer, gas separation system.
Atmospheric Structure Instrument	3.00	12.00	(2b/s)	200×200×200	In situ measurements of atmospheric density, pressure, temperature profile. Huygens/HASI heritage
Helium abundance detector	1.20	1.20	(4b/s) 1 sample/64 sec	TBD	Measurement of He abundance in Neptune's atmosphere; flown on Galileo Probe.
Radio Science Experiment (USO)	1.80	3.60	0.055	40 Ø × 140	Ultra-Stable Oscillator to generate a stable signal for the Probe radio link.
Nephelometer	2.76	3.60	0.15	TBD	For cloud locations and aerosol properties
Net-flux radiometer (NFR)	2.88	7.56	0.06	110×140×280	Measure the net radiation flux and upward radiation flux within the atmosphere. Heritage: Venus Probe/LIR and Galileo Probe/NFR.

Table 6-4: Augmented Atmospheric Probe Model Payload

The total mass of this Model Payload amounts to 30.6kg, including 20% maturity margin and an average power consumption of ~109W (no margin included).

With further payload iterations, the SST is confident to be able to reduce the P/L mass to ca. 20kg (incl. maturity margin) without compromising science.

6.1.5.1 Payload Components of the augmented Atmospheric Probe Model Payload

More details on some of the instruments are listed here below:

- **Mass Spectrometer:** The time-of-flight MS consists of four units: the MS itself, a tuneable laser spectrometer, a gas separation and enrichment system, and the reference gas system.
- **Nephelometer:** The instrument would passively sample cloud and haze particles, illuminate them, and measure the flux and degree of polarization of the scattered light. The TRL for light-weight designs (1kg, <3W) is TBD. The instrument contains two modules: LOAC (Light Optical Aerosol Counter) to measure the size distribution of particles, and PAVO (Polarimetric Aerosol Versatile Observatory) to measure particle shape and composition. It was flown on balloons in Europe (LOAC-S instrument). If LOAC only is considered, the mass can be reduced to <1 kg.

6.1.6 Technology Needs

No new technologies were identified for the baseline Model Payload (Table 6-3).

6.2 Orbiter

6.2.1 Requirements and Design Drivers

The Neptune Orbiter Model Payload is based on the science objectives and measurements described in the Science Traceability Matrix (STM) for Neptune, as defined by the Study Science Team (SST).

The Model Payload was put together to size the spacecraft resources and to allow for accommodation checks. These instruments are a representation of a possible future payload, but are not meant to be understood as a pre-selection of instruments for a potential future mission.

Currently only the data volume produced by the science instruments was identified to impact the Mission/System requirements. The SST has established preliminary observation time-lines [see Table 5-18] for different parts of the science observations that support the determination of the communications system.

Several instruments are nadir pointing (similar pointing accuracy requirements as for the JUICE mission were provided) and no EMC requirements have been established at the time of the study. The payload accommodation and access of the instruments to their measurement environment needed to be considered for the baseline design.

6.2.2 Assumptions and Trade-Offs

Assumptions	
1	Power for science operation shall be provided by (e)MMRTGs (see 0).

6.2.3 Baseline Design

The list of Model Payload instruments is given in Table 6-5. It shows the instruments' mass, power and data rate as well as the volume envelope. Note that the mass values already include 20% equipment margin. The Notes/Heritage column contains information on precursor instruments.

Instrument	Mass [kg]	Power [W]	Data rate [kb/s]	Volume envelope [mm]	Notes/Heritage
Camera	19.20	34.80	550.00	660×490×300	Simplified JUICE/JANUS design; Narrow-angle framing camera, 13 filters (TBC), spectral range 350-1050 nm
Imaging Spectrometer	18.60	25.20	2870.00	500×550×250	Simplified, single-channel JUICE/MAJIS design; spectral range 0.4-2.5 microns
Ion and Neutral Mass Spectrometer	43.20	21.24	1.30	260×260×170 630×630×260	Rosetta/ROSINA design; two mass spectrometers (DFMS, RTOF), one pressure sensor

Instrument	Mass [kg]	Power [W]	Data rate [kb/s]	Volume envelope [mm]	Notes/Heritage
(3 parts: COPS, DFMS, RTOF)				380×1140×240	(COPS). Similar to Cassini/INMS
Magnetometer	5.47	12.00	1.20	10000 (boom length) e-box: 300×200×200 fluxgate sensor: 110×110×120 (each)	Fluxgates sensors mounted on boom; JUICE/J-MAG design
Microwave radiometer	23.21	67.14	5.23	550×392×451	Based on JUICE/SWI; Wavelength range: 1.37-50cm (=600 MHz-22 GHz); for deep atmosphere
Radio Science: - USO (Ultra-Stable Oscillator) - X/Ka-band Transponder	2.40 4.13	6.00 39.84	0.00	172×154×118 236×208×150	Radio science package: X/Ka-band transponder + Ultra-stable oscillator for the gravity science and radio occultations, both at the Ice Giant and the satellites. BepiColombo/MORE and JUICE/3GM heritage.

Table 6-5: Model Payload for the Neptune Orbiter

The total mass of this Model Payload amounts to 116.2kg, including 20% maturity margin. The average power consumption of this payload is ~248W (no margin included).

In Section 6.2.3.1 more details on the assumed masses and the derived data rates of the Model payload instruments are given. In addition, modifications proposed with respect to the existing/heritage instruments used to derive the Model Payload for this study are listed.

6.2.3.1 Payload Components

- **Camera:** A single, uncompressed acquisition is 46 Mbit and the conservative compression factor is 3-3.5. A realistic compression factor could be of 7 (BepiColombo-SIMBIO-SYS currently adopted compression factor for HRIC) but values up to 28 are possible with higher image degradation.
- **Imaging Spectrometer**: A single, uncompressed acquisition is 8.6 Mbit and the expected compression factor is 3. The JUICE-MAJIS data rate was halved for this instrument, as only a single channel is use for this light version. This amounts to 2870kb/s. The power for this simplified design is the JUICE-MAJIS power need, but scaled down by 25%.
- **Ion and Neutral Mass Spectrometer:** The mass of 36 kg and power of 42W comprises all three instrument elements together. The data rate of 1.3 kb/s is based on "nucleus mapping" of the Rosetta-ROSINA instrument and 25 b/s is allocated for house-keeping.

- **Magnetometer:** The mass for the magnetometer is based on JUICE-J-MAG, but lowered by 7% to compensate for the radiation shielding that was added to J-MAG. The boom length is also based on JUICE-J-MAG. The EMC requirements discussion is pending, as this depends on the S/C EMC environment. The data rate is taken from the J-MAG normal mode of 2.41kb/s and it is already compressed. Also, 16 vectors/s could be used for the Ice Giants mission instead of 32, which halves the data rate to 1.2 kb/s. The electronics box is mounted on the service module of the S/C, two fluxgate sensors are mounted on a boom (one of the sensors at the end of the boom), each is sensor connected to the box by a harness. The electronics box volume is 300×200×200 mm, and each fluxgate sensor has a volume of 110×110×120 mm. The boom length is 10000 mm.
- **Microwave radiometer:** Note that the warm-up power for this instrument is 24.46W for 60 min. The dimensions are in detail 550×392×451 mm for the Telescope and Receiver Unit and 489×489×40 mm for the Radiator.
- **Radio Science:** This instrument package contains a X/Ka-band transponder and Ultra-Stable Oscillator for the gravity science and radio occultations, both at the giant planet and the satellites. Note that this X/Ka-band transponder is in addition to the platform communications Ka-band transponder. The produced data rates are limited and the radio science measurements plus housekeeping telemetry is estimated to be in total ~10% of the data volume of the other instruments. The transponder needs warm-up power of 35.7W for 5 min.

6.2.4 List of Equipment

See Table 6-5.

6.2.5 Options

No Options were studied.

6.2.6 Technology Needs

No new technologies were identified for the baseline Model Payload (Table 6-5).

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7 NEPTUNE CONFIGURATION

7.1 Atmospheric Probe

7.1.1 Requirements and Design Drivers

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
CONF-010	Keep same baseline design as PEP-V (SUN) probe	
CONF -020	Assign new mass and volume for varied equipment	
CONF -030	Scale the descent module with 10 [cm] in diameter	
CONF -040	Keep pressure vessel dimensions intact	

7.1.2 Assumptions and Trade-Offs

For the probe configuration, the same baseline design was kept as depicted in previous CDF studies, the PEP-V probe, mission to Venus, and the PEP-SUN probe, mission to Saturn, Uranus and Neptune, both conducted in the CDF during the summer of 2010 RD[7]. Thus, the configuration was adapted for this new mission, the M* Ice Giants Study, mission to Uranus and Neptune.

The probe design consists of a descent module, which is a spherical pressurised vessel accommodating the payload and equipment, and a deceleration module made out of two parts, a front and a back shield. These shields have the same objective, to protect the pressure vessel from the extreme heat loads during its decent.

As mentioned above, one of the major requirement of the probe configuration was to keep the same baseline design of the PEP-V probe, driven by aerodynamic constraints, but to enlarge the deceleration module diameter from the base diameter of PEP-V design of 1250 [mm] to a new diameter of 1350 [mm]. The descent module should remain unchanged, dimension wise, accommodating all the subsystem units.

By scaling the front and back shield of the probe, the TPS design was consequently altered. The EDS subsystem was modified as well, and a new, bigger volume was needed to be allocated below the back shield.

7.1.3 Baseline Design

Based on PEP-V design, the internal accommodation in the decent module pressure vessel was retained, except the following:

- Batteries size and redistribution
- New solid state power amplifiers (SSPA) and new patch antenna design
- Data handling components changed
- Reshuffling of the components inside the descent module.

The location of the helix antenna, that supports the data link with the orbiter, was lowered as much as the available volume permitted, to make space for the new EDS design. The new available envelope dedicated for the EDS subsystem is of 0.05 [m^3].

As needed, the deceleration module was lengthened with 10 [cm] and the new TPS thickness were applied calculated by the thermal subsystem experts (Figure 7-8; BS employs the same design outline).
Under these circumstances, with a FS-BS thicker, the location of the pressure vessel needed to be raised as well.

7.1.4 Overall Dimensions

The final design's overall dimensions of the M* Ice Giants Study probe are shown in the figures below (Figure 7-3 & Figure 7-7), and the internal accommodation of the units for the aft and forward compartments are shown in Figure 7-4 and Figure 7-5 respectively.

The CoG of the entry probe given by CATIA is listed below (Table 7-1) and calculated from the nose area.

COG	
Gx	2.208 [mm]
Gy	-1.018 [mm]
Gz	500.698 [mm]

Table 7-1: CATIA output for M* Ice Giants Study for probe

For more detailed information about the probe configuration, please refer to the document describing the PEP-V probe configuration RD[7].

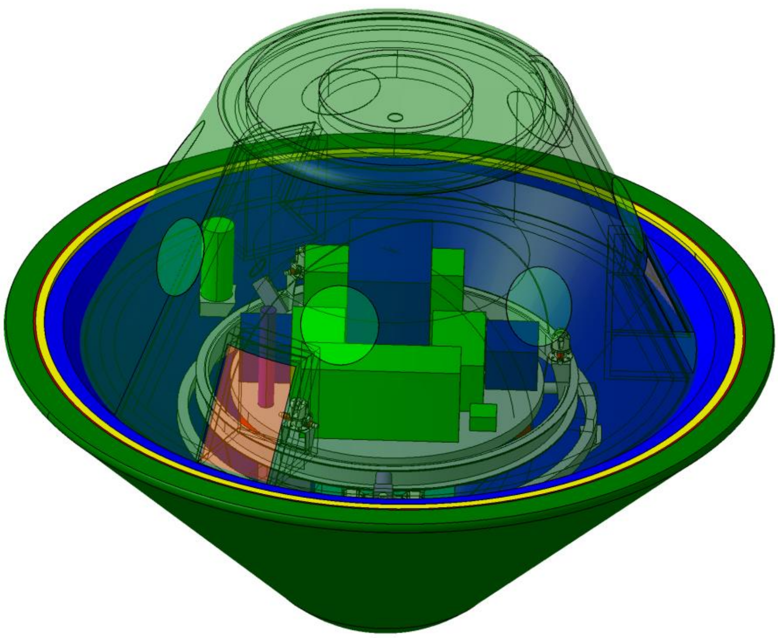

Figure 7-1: M* Giants Study probe configuration

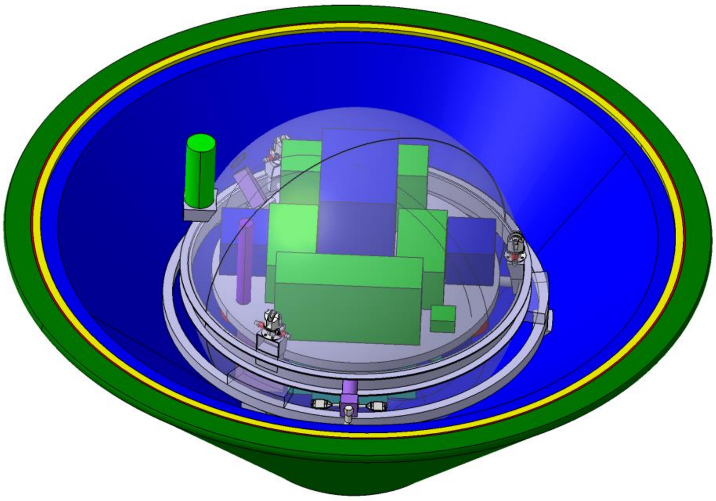

Figure 7-2: M* Giants Study probe config –view descent module

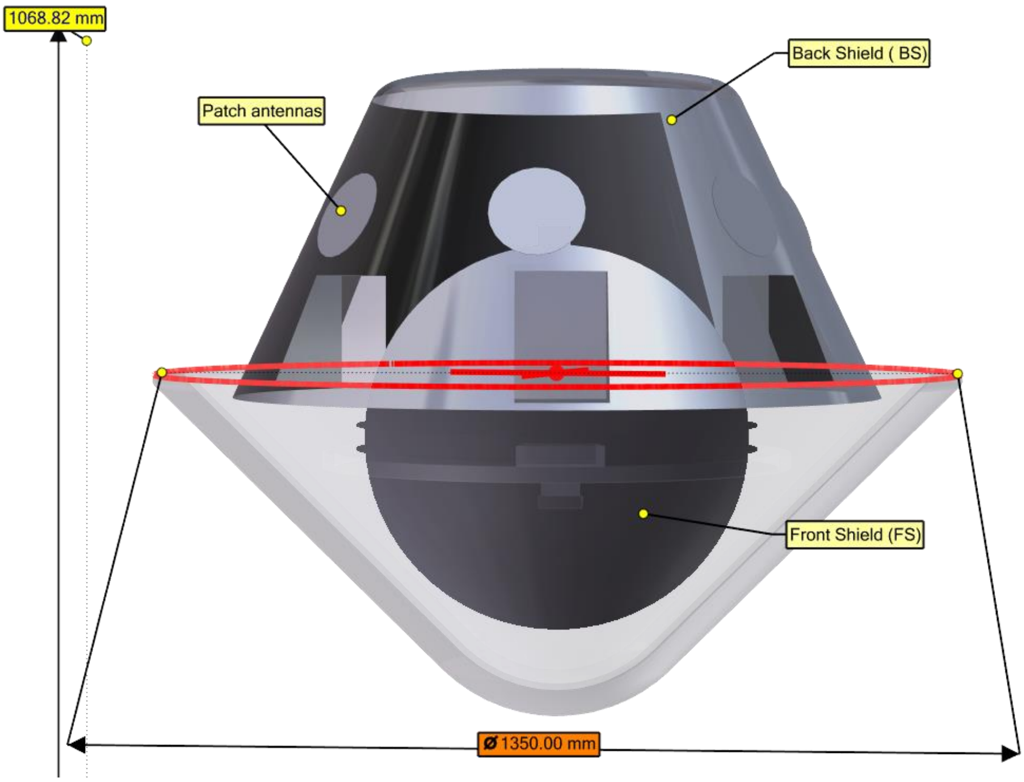

Figure 7-3: M* Giants Study probe config –overall dim.

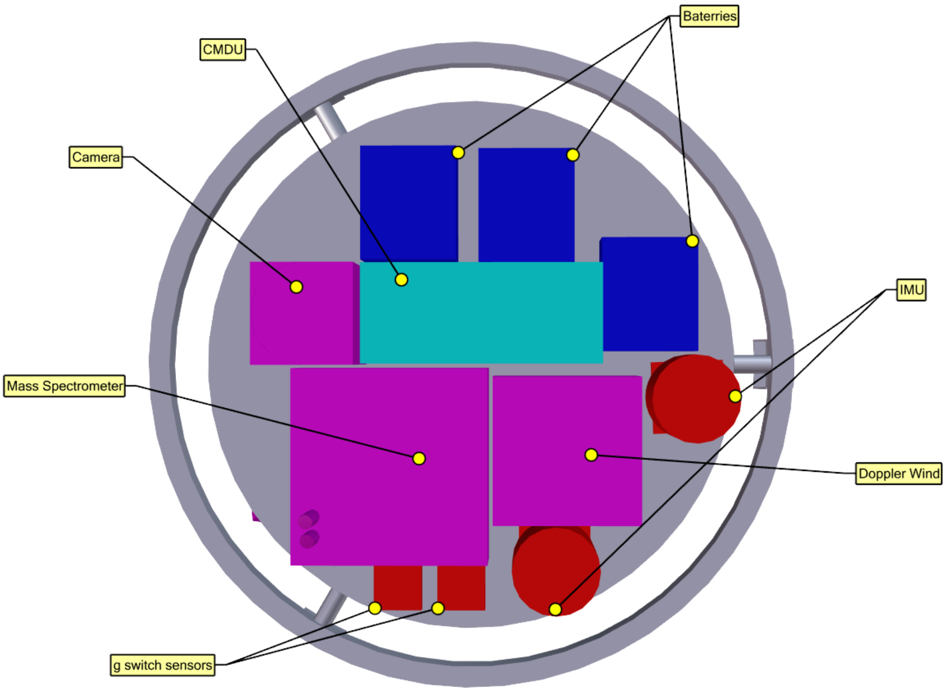

Figure 7-4: Descent module – accommodation top platform

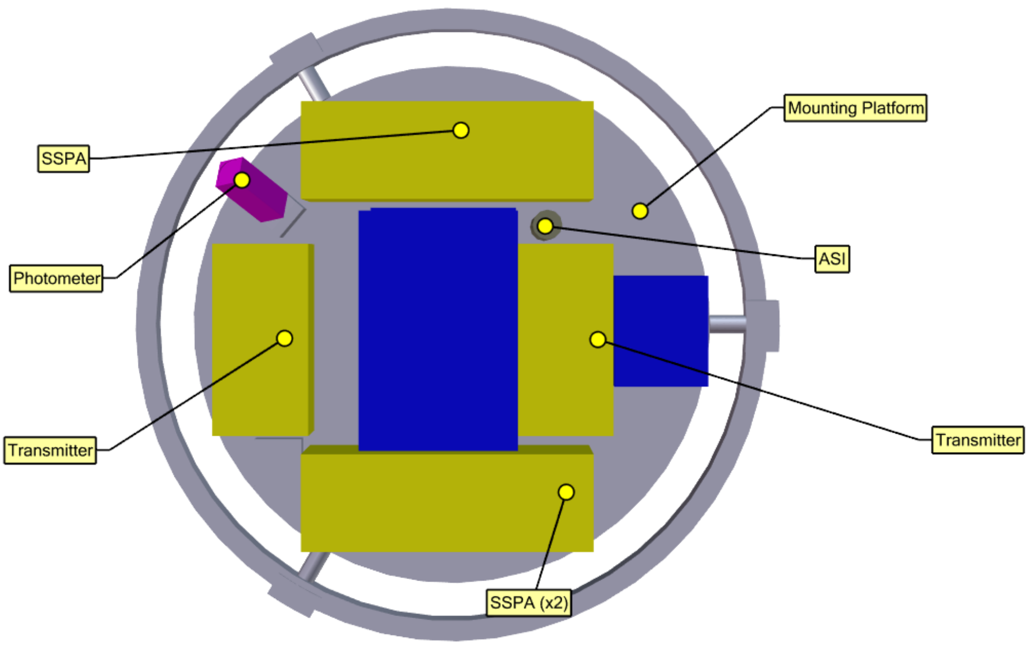

Figure 7-5: Descent module – accommodation bottom platform

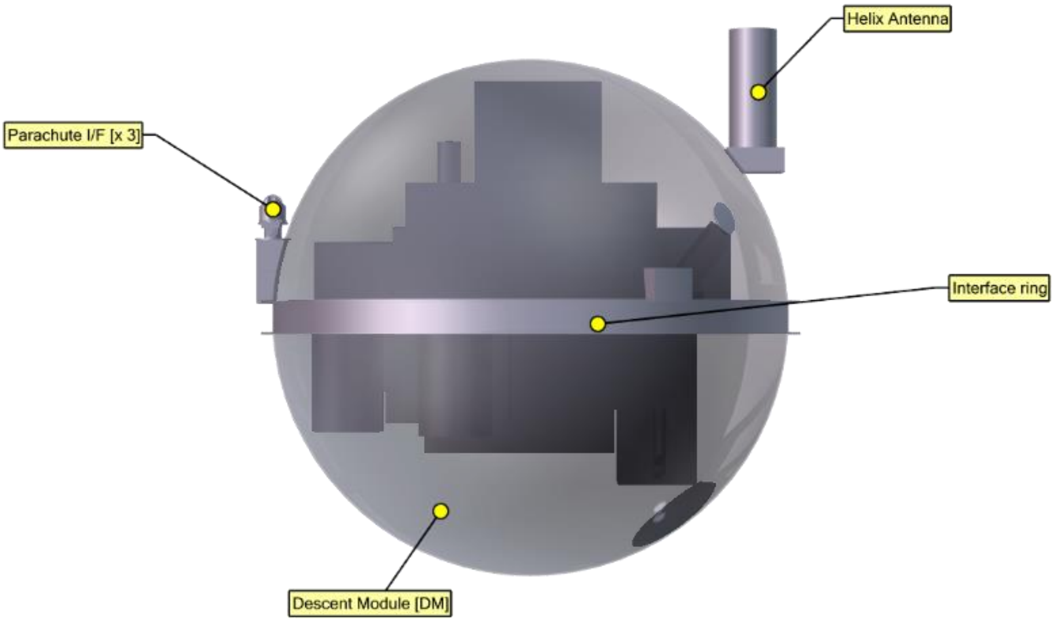

Figure 7-6: Descent module

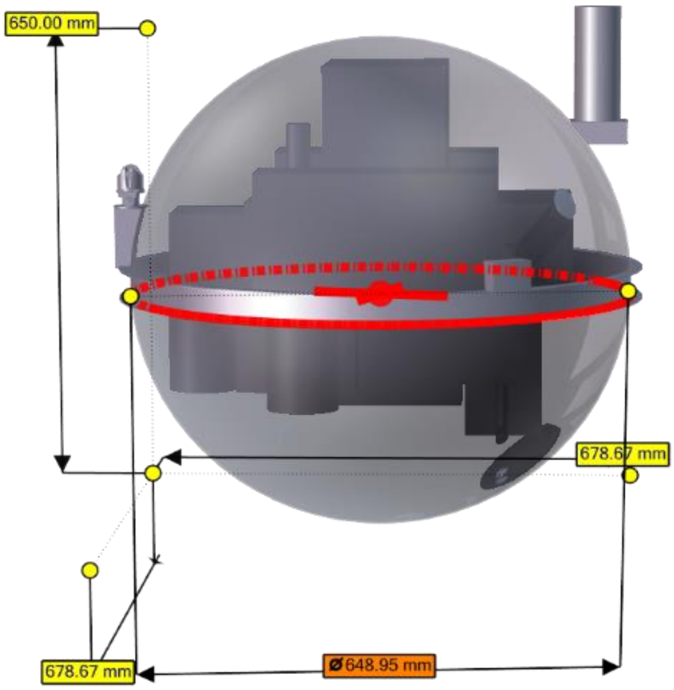

Figure 7-7: Descent module –overall dimensions

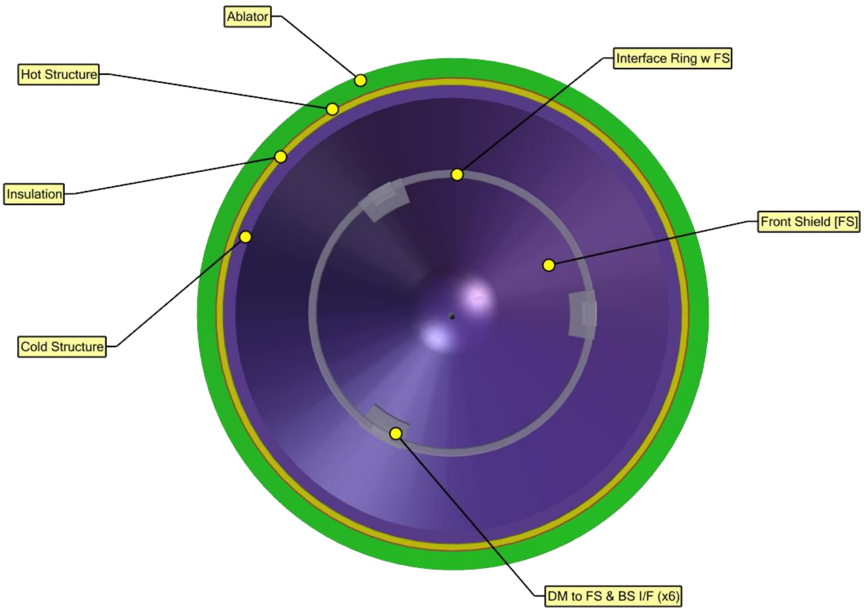

Figure 7-8: Front Shield –TPS design

7.2 Orbiter

7.2.1 Requirements and Design Drivers

The spacecraft shall provide accommodation for all the sub-systems and ensure the required pointing, if any mentioned.

7.2.2 Assumptions and Trade-Offs

Assumptions	
1	Mass implemented in CATIA with a 20% system margin on top of the 20% at product level
2	If no physical representation of the component/part is available, mass gets redistributed accordingly, to its own sub-system, or uniformly to the overall mass of the spacecraft

7.2.3 Baseline Design

The configuration was driven largely by the propulsion subsystem, which provides the necessary thrust and manoeuvres for such a long duration mission. Four bi-propellant tanks main engine were selected from the E3000 tank family along with three helium, pressurizing tanks. These were positioned as much as possible around the centre tube of the orbiter configuration.

In addition to the centre tube used for reinforcement, shear panels were added for accommodating the instruments, bottom and top panels, and of course, panels that enclose the overall configuration.

In order to isolate the instrumentation, a dedicated panel was reserved on which an optical bench will be mounted with standoff on the outside panel of the spacecraft.

Thus, the optical bench will be isolated from the orbiter and located near the upper end of the spacecraft. Along with the orbiter's payload, the two NavCams were also positioned on this optical bench,, as they are required to point in the same direction as the camera. The standard radiator was placed under the optical bench to facilitate the heat exchanged in the hot case from the payload instruments, and a louvered radiator in close proximity for the same reason.

Other major equipment were the three enhanced Multi-Mission Radioisotope Generators (eMMRGT), which were placed near the bottom end, on the outside of the spacecraft, and in their close proximity, the excess power radiator.

Furthermore, the four space large-format Li-Ion batteries of 44 kg each were placed as well internally at the bottom of the spacecraft. The comms high gain antenna, of 3[m] in diameter was fixed on the top deck of the spacecraft (configuration during launch) and the 10 [m] unfolded boom, with the magnetometer at the far end, was positioned on one side of the spacecraft.

For the AOCS subsystem, the four reaction wheels were placed as close as possible to the centre tube and the star trackers were positioned 180 degrees opposite to the optical bench, as required. S10-18 thrusters will provide the attitude control of the orbiter and desaturation of the reaction wheels as depicted by the propulsion subsystem. These do not have a physical representation in the current configuration.

All other subsystems, from power, data handling, communication components, sensors and gyros of the GNC subsystem, were as much as possible grouped together on the shear panels, in order to facilitate the need of having an extra structure protecting them from the radiation environment.

The orbiter configuration can be seen in Figure 7-9 and the (internal) configuration of the subsystems and units labelling is shown in Figure 7-10 and Figure 7-11.

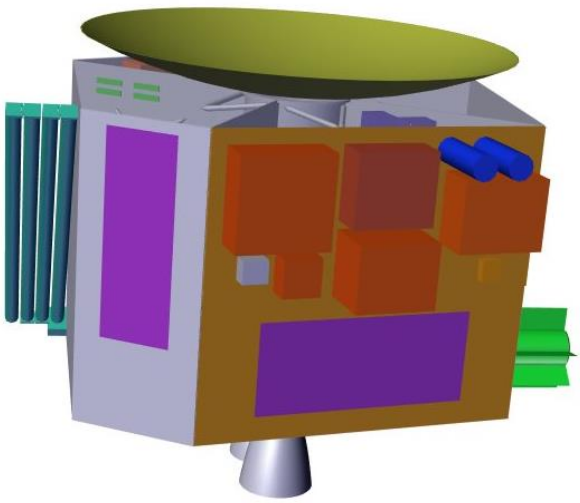

Figure 7-9: Orbiter configuration

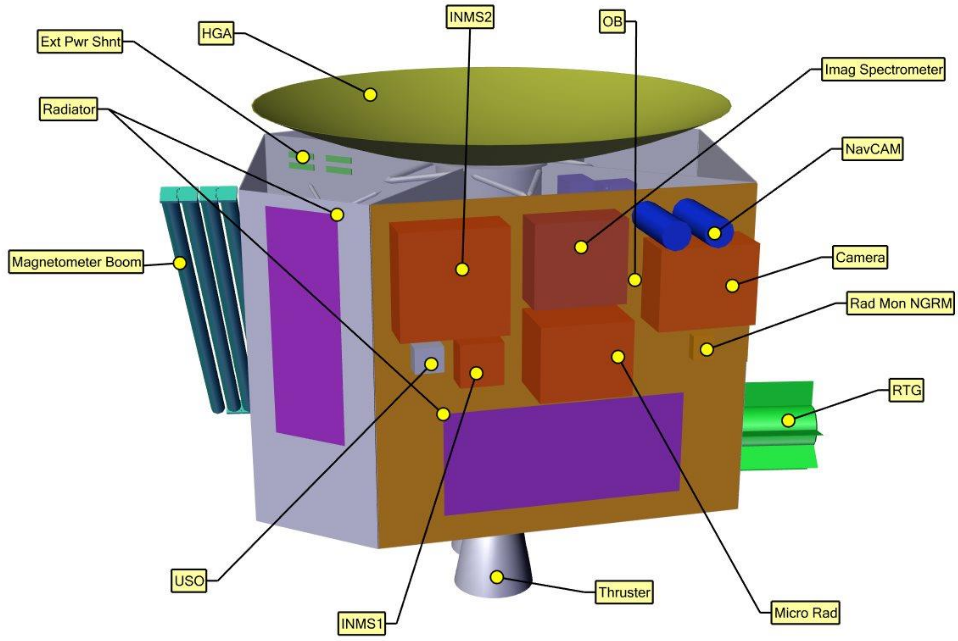

Figure 7-10: Orbiter equipment labelled

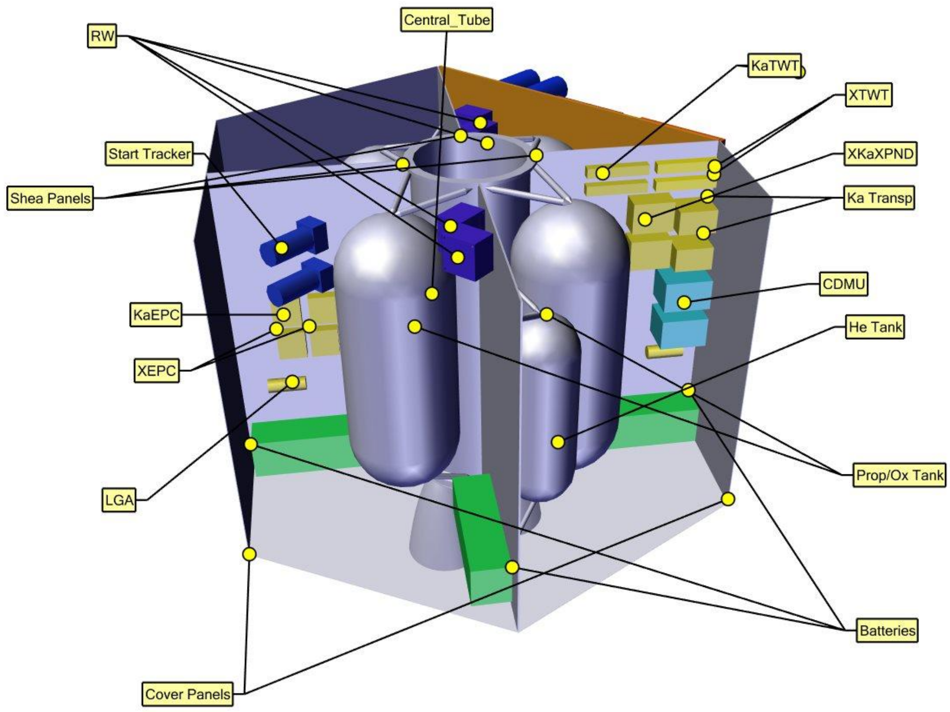

Figure 7-11: Orbiter internal equipment labelled

7.2.4 Overall Dimensions

Overall dimensions of the M* Giants Study orbiter to Neptune can be visualised in Figure 7-12.

The total dry mass of the Neptune orbiter is 1603.4 kg and the position of the centre of gravity, as output by CATIA, is shown in Table 7-2.

COG	
Gx	-5.796 [mm]
Gy	9.606 [mm]
Gz	1074.793 [mm]

Table 7-2: CATIA output for M* Ice Giants Study for orbiter

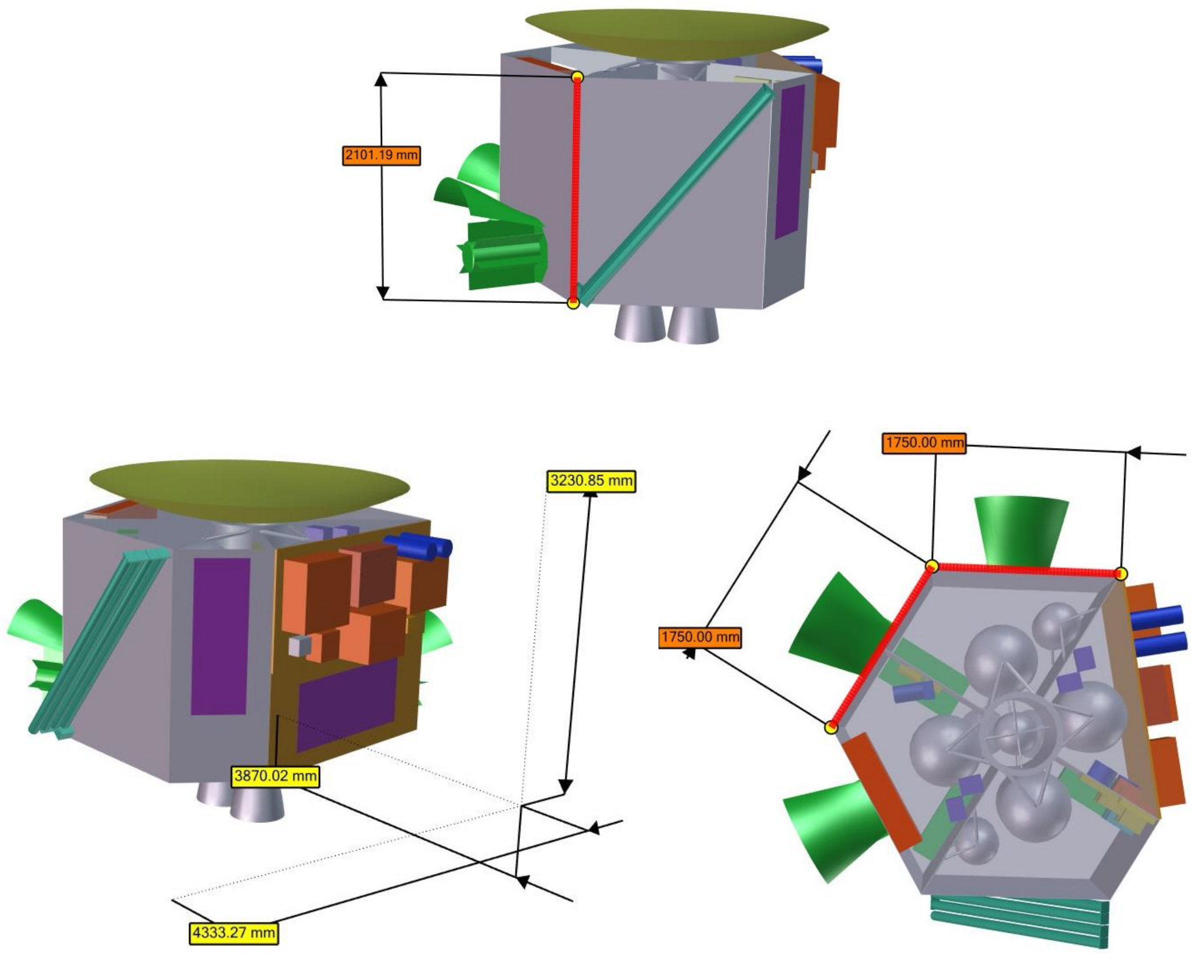

Figure 7-12: Orbiter –overall dimensions

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8 NEPTUNE STRUCTURES

8.1 Atmospheric Probe

8.1.1 Requirements and Design Drivers

The heritage of the Probe mechanical design is sourced in the design concept of the Planetary Entry Probe for Venus, (PEP-V), assumed to be a baseline for outer Solar Entry Planetary Probes for Saturn, Uranus and Neptune missions, RD[7]. The reason for this assumption is in the fact that PEP-V represented the worst case environmental conditions (atmospheric pressure) hence enveloping all above mentioned mission environmental requirements. Whilst there were some configuration changes, such as; a variation in the TPS thickness applied, number of parachutes used (Venus required a drogue and main chute, whereas Saturn, PEP-Neptune and Uranus required only a drogue chute); the structural requirements and loads were assumed to remain unchanged. For the Ice Giants study, the approach was to take the PEP structural design as reference and adapt it in accordance to the current pressure requirements (10 bar).

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
STR-20	The Neptune probe shall accommodate and operate the scientific P/L, avionics and power subsystems in a descent module compatible with atmospheric conditions to an altitude corresponding to at least 10bar	
STR-25	In design of the Neptune probe the following mass margins shall be used: • Conventional maturity margins for all subsystems, between 5 and 20%, depending on the maturity level agreed with Agency • A system margin of 20% on top of all equipment, except for the TPS material (back and front). The heat shield mass will be computed using aerothermodynamics data, including their margins and based on the NEP mass including margins (and heat shield mass) as defined above • A 50% maturity margin shall be added to the mass of the heat shield material computed as specified above if the current TRL is lower than 5	
STR-30	Max. deceleration shall not exceed 50 g's	
STR-35	Max. front shield, (FS) pressure shall not exceed 10.2 bar, resulting in the load of 140 kN	

From the list of requirements, the following requirements are identified as design drivers and will be followed by the detailed design assessment presented in this report.

- Descent Pressure – driving descent module, (DM) wall thickness and hence mass. It will be investigated how the atmospheric pressure affects the design of the DM shell thickness. The assessment presented here follows analytical approach to determine DM minimal shell thickness able to sustain external pressure, as explained in RD[12]

- Entry Decceleration – driving front shield pressure and hence loading on Front shield and loads at front shield/Descent Module (DM) interface.

8.1.2 Assumptions and Trade-Offs

Assumptions	
1	For simplicity. A perfect spherical shape of the descent module (DM) shell is assumed, without any cutouts and/or reinforcements. DM pressure shell material is Titanium, with diameter of 650mm.
2	Design of the FS cold structure is based on the Al honeycomb core, with CFRP skin. The core shall support inserts that will transmit more than 10kN in shear.
3	The alternative is to have a monolithic structure, either metallic, CFRP or potentially modification of the C/SiC of the TPS, able to sustain greater loads.
4	No FEA has been conducted in this Study, however it is essential for future work in order to assess structural strength around interfacing structures/ports/inlets and hence better mass estimation.

8.1.3 Baseline Design

The Neptune probe structural design is based on the Venus PEP design. Internal accommodation of the DM spherical pressure vessel was assumed the same as the one used for the Venus case. The primary structure of the Probe contains two elements: Front and Rear Shields, which are forming aerodynamic element to ensure initial entry and descent, and DM shell, which accommodates instruments and payloads and essentially is defined as a pressure vessel. In addition to these, there are also secondary structural elements such as:

- Payload mounting platform that is situated equatorially in the Descent module and has a diameter of 630mm. Payload mounting platform is a standard CFRP/Aluminium core sandwich panel, 20mm thick with 0.6mm CFRP skin. All equipment and payload are mounted on both sides of this panel, which in turn is mounted to the DM connection ring.
- Front shield interface brackets, which interface the Front shield with the separation system connected to the DM connection ring
- Back Shield to DM interface bracket, connecting the back shield separation system to the DM
- Back Shield ribs, these act as supports for the mortar on the back shield
- DM main parachute support structure provides support to the mortar and parachute for the Descent Module
- Miscellaneous structural items are inserted to cover items not detailed in the current design.

The assumptions given in section 8.1.2 and corresponding configuration assessment of the instruments and payload mounted on the DM platform as done in the course of Study foresees following parameters for the DM shell:

- Material to be used: Titanium
- Diameter of the DM pressure shell: 650mm
- Max. external pressure: 10bar.

Based on the input data above, the shell thickness has been determined, based on the inversion of the following expression defined in RD[12]:

$$q^{'} = \frac{2E\,t^2}{r^2\sqrt{3\left(1-\upsilon^2\right)}} \qquad \textbf{Equation 1}$$

Where $q^{'} = \frac{2E\,t^2}{r^2\sqrt{3\left(1-\upsilon^2\right)}}$ Equation 1 can be simplified as follows:

$$q^{'} = \frac{0.365E\,t^2}{r^2} \qquad \textbf{Equation 2}$$

The $q^{'} = \frac{0.365E\,t^2}{r^2}$ Equation 2 uses knock down factor for material anomalies in the range between 0.365 and 0.840 in ideal case. In the course of PEP-V study, it was agreed to consider value of 0.5 as a realistic one and to allow some mass savings. The same value has been assumed for the Ice Giants study. Re-writing $q^{'} = \frac{0.365E\,t^2}{r^2}$

Equation 2 to obtain thickness as a function of pressure q', geometry (radii r) and material moduli of elasticity, E, yields to:

$$t = \sqrt{\frac{r^2 q^{'}}{0.5E}} \qquad \textbf{Equation 3}$$

Applying simultaneously qualification factor of 2 and buckling factor of 1.25 on the pressure load, considering radii of DM shell to be 325mm and E=114000 N/mm² for Titanium: finally, shell thickness of: 2.2mm is obtained. Considering DM shell surface area and calculated thickness for the Titanium material, the mass of the shell is calculated to be 12.0kg.

8.1.4 List of Equipment

Based on the current configuration of the Probe and considering similar technological advances and material characteristics of the main structural parts identified in Sec.8.1.3, the following mass budget as presented in Table 8-1 is obtained.

Item	Nr	Item mass, kg	Structure mass	Material	Maturity	Margin %	Unit mass with margin
FS – cold structure	1	7.50	7.50	CFRP/Al-core	New develop.	20	9.00
FS – I/F bracket	3	1.32	3.96	Ti	New develop.	20	1.58
BS – cold structure	1	4.05	4.05	CFRP/Al-core	New develop.	20	4.86
BS to DM I/F bracket	3	1.32	3.96	Ti	New develop.	20	1.58
BS stiffening ribs	3	1.00	3.00	Ti	New develop.	20	1.20
DM upper shell	1	6.00	6.00	Ti	New develop.	20	7.20
DM lower shell	1	6.00	6.00	Ti	New develop.	20	7.20
DM mid. Section ring	1	5.70	5.70	Ti	New develop.	20	6.84
DM connection ring	1	4.79	4,79	Ti	New develop.	20	5.75
DM mounting platform	1	1.03	1.03	CFRP/Al-core	New develop.	20	1.24
Main parachute support	3	1.68	5.20	Ti	New develop.	20	2.02
TOTAL:			**51.03**			**20**	**61.24**

Table 8-1: Probe list of equipment and mass budget breakdown

8.1.5 Options

Currently no further design options for the Probe have been investigated in the course of this study. However, in the course of further development of the Probe mechanical design it is important to emphasise the importance of detailed structural analyses of the front shield interfacing structure in order to optimise mass by reducing density of the front shield core by introducing larger number of inserts. For this purpose detailed FEM of the front shield and interfacing structures would be necessary in the next stage of this study.

8.1.6 Technology Needs

No new technologies are identified at this stage of the study.

8.2 Orbiter

The structures subsystem mechanical concept is conceived throughout configuration assessment in order to provide efficient support and accommodation to the major load contributors, e.g. propulsion subsystem, based on the bi-propellant concept of four E3000 tank family, together with three pressurant tanks (He), as well as providing support and accommodation for other spacecraft subsystems: payload, power, communications, thermal, AOCS, etc. The launch concept follows NASA dual stack configuration above SEP, as discussed in the RD[1] and further assessed in the course of ESA CDF study.

8.2.1 Requirements and Design Drivers

The assessment of the mechanical design drivers follows the logic of the mission launch scenario based on the SLS as selected launch vehicle and launcher's environment predictions given in RD[13], which shall be used in payload structural design. These predictions assume dynamic excitations, occurring predominantly during lift-off and transonic periods of SLS flight and are superimposed to the steady-state accelerations to produce combined accelerations (expressed in g's) in the Table 8-2.

Event		Lateral Accelerations	Axial Accelerations
Liftoff	min	0	-1.5
	max	2	-1.5
Ascent - Transonic	max	2	2.25
Booster Phase - Max G	max	0.5	3.25
Core Stage Phase - Max G	max	0.5	4.1

Table 8-2: Quasi-static Loads for SLS Flight Conditions

Furthermore, in the course of Study the following structural requirements have been used as design drivers (mass and stiffness driven design) for the mechanical concept of the Orbiter:

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
STR-010	The overall mass budget for the s/c structures subsystem shall not be higher than 300kg (TBC), including design maturity margin of 20%	
STR-020	Cantilevered payload fundamental mode frequencies are assumed to be min. of 8Hz lateral and 15Hz axial to ensure applicability of the design load factors for the QSL as mentioned in Table 8-2	

8.2.2 Assumptions and Trade-Offs

The launch scenario foresees usage of SLS 8.4m long P/L fairing, under which two spacecraft in stacked configuration shall be mounted on the NASA SEP module. The assumptions and trade-offs conducted in the course of study are summarised in the following table:

	Stacked	Side-by-side
Design and interface requirements	-/+ Might be challenging for SEP supporting structure and interfaces	+/- (lower loads on SEP, however due to large lateral bending possible increase of Orbiter's structural mass)
Delta-v and deployment requirements	-	+
Stability	+/- (both Orbiters shall be well balanced)	-
Launch envelope	0	0
Previous experience	++	-

Table 8-3 Launch Configuration Trade-offs

The outcome of the above conducted trade-offs resulted in two orbiter spacecraft stacked in the LV axis direction above SEP module as a most suitable configuration, based on the previous missions designing experience, (e.g. Bepicolombo). Such configuration ensures potential for the orbiters structures mass reduction as a design driver, however at the same time puts some challenges in design of the SEP supporting structures and stack I/F points. The accommodation of the stack configuration under 8.4m long payload fairing is shown in Figure 8-1.

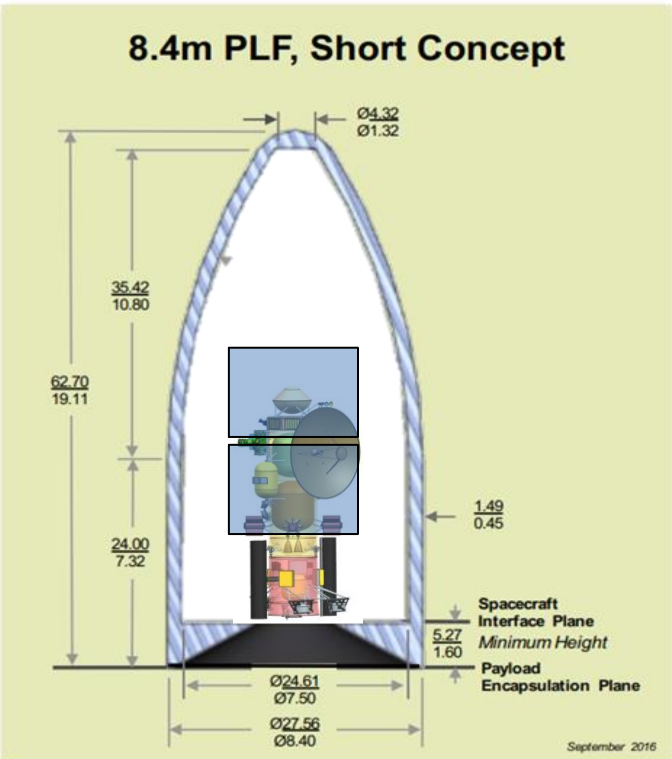

Figure 8-1: Stacked Orbiters accommodation under 8.4m PLF

Based on the transfer-to-Neptune mission scenario and corresponding limitations to the bi-propellant based propulsion of Orbiters with assumed capacity of the E3000 tanks volume, the following design assumptions have been taken:

Assumptions	
1	Assumption 1: overall structures subsystem mass for both orbiters shall not exceed 600kg (TBC), including mass of stack I/Fs and supporting structures for the high gain antenna and 10m long magnetometer boom
2	The general shape of stack is based on the cylindrical bus, with shear and side assembly panels, top and bottom floors for accommodation of the s/c subsystems, payload and equipment.

8.2.3 Baseline Design

Based on the above-defined assumptions, design drivers, and Neptune Orbiter configuration assessment, the following s/c structure is shown in Figure 8-2 below:

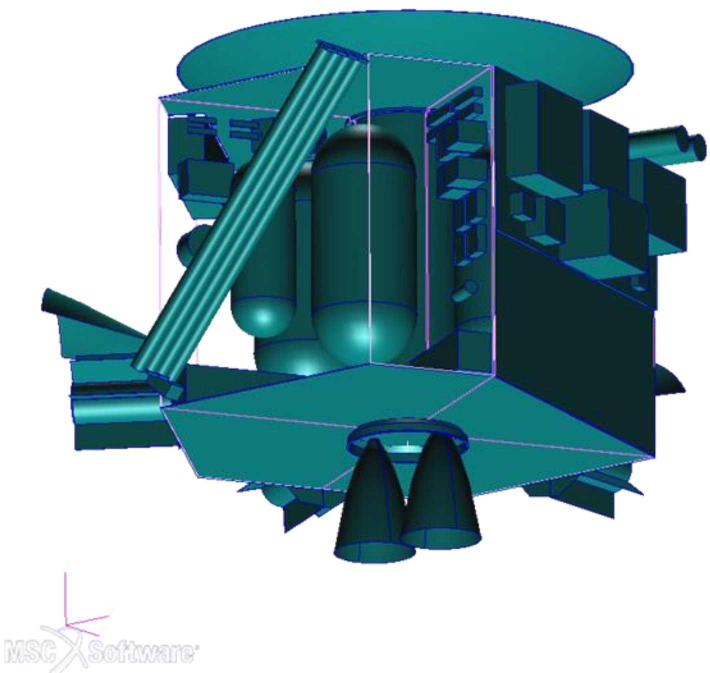

Figure 8-2: Neptune Orbiter structures subsystem

Transparent view of the side assembly panels in the figure above shows four core structures of shear panels around tubular support, (CFRP filament-based cylinder). All panels are sandwich panels of 20mm Al-honeycomb, with 0.6mm CFRP skin. The tubular element accommodates pressurant tank and provides propellant tank supporting trusses interfaced together with shear panels thus ensuring transfer of the major mass inertia loads via stack interfaces to SEP module mounted on the LV adapter.

8.2.4 List of Equipment

The list of structural subsystem parts/equipment is extracted from OCDT and shown in Table 8-4:

	mass (kg)	mass margin (%)	mass incl. margin (kg)
SC (Spacecraft)	252.72	20.00	303.26
APs (Assembly Panels)	54.82	20.00	65.78
BP (Bottom Panel)	18.16	20.00	21.79
CPROP_TD (CPROP_Tank Deck)	21.88	20.00	26.26
MC (Module Collars)	22.00	20.00	26.40
SPs (Shear_Panels)	28.64	20.00	34.37
TP (Top Panel)	18.16	20.00	21.79
TR (Tube Rings)	12.74	20.00	15.29
TSS (Tank Supporting Struts)	66.00	20.00	79.20
TST (Tank Supporting Tube)	10.32	20.00	12.38

Table 8-4: Orbiter structures subsystem mass breakdown

The mass breakdown for the primary and secondary structural elements includes system maturity margin adopted in this study of 20% and shows slightly increased total mass above requirement STR-010 but still within acceptable tolerance, with regards to required propellant mass and delta V characteristics.

8.2.5 Options

It has been clearly stated in the course of the study that stiffness verification of such complex stack configuration of three modules (two orbiters and SEP module) would require detailed FEM analysis, which overcomes scope of work in this study and accessibility to the necessary information about SEP at this project phase. However, such analysis is required to assess stiffness characteristic of the proposed configuration against the requirement STR-020 in order to justify design limit load factors adopted for SLS. Demonstration of the stiffness compliance in this case is also necessary in order to ensure proper dimensioning of the primary structural elements, as well as interfacing structures to the major system masses in the next project phase.

8.2.6 Technology Needs

No new technologies are identified at this stage of the study.

9 NEPTUNE MECHANISMS

9.1 Atmospheric Probe

9.1.1 Requirements and Design Drivers

The following tasks have to be covered by mechanisms in the probe.

- Separation from the spacecraft and stabilisation via spin. This task is covered by a spin and eject mechanism. A separation speed of 0.4m/s and a lateral speed lower than 2.5cm/s are required. A spin of 2-3rpm is considered for now, similar to previous missions (i.e. ExoMars). Connection to the spacecraft is done at 3 points, thus synchronised actuation of the three release mechanisms is required.
- Front shield separation. The front shield is separated from the rest of the probe when no longer needed. The shield is connected to the rest of the probe in three points, which have to be actuated simultaneously.
- Back cover separation. The back cover is separated when needed by three simultaneous release separation mechanisms. The back cover is attached to the descent module in three points to be released simultaneously.
- Cut cable line connecting to mortar-released pilot parachute. A cable cutter has to be included in the design to cut the connection to the mortar-deployed pilot parachute.

9.1.2 Assumptions and Trade-Offs

The previous CDF Study PEP (Planetary Entry Probe), as well as the design of the Huygens probe, are taken as reference.

The mechanisms related to parachute deployment and parachute swivel are covered by the parachute discipline.

9.1.2.1 Trade-off for release actuators

The mechanisms hold the separating parts together until the time of separation. This separation is initiated by release actuators. Several possible technologies have been considered for release actuators: pyrotechnic actuators, non-explosive actuators (NEA), shape memory alloy based actuators.

NEAs and shape memory alloy actuators provide a lower shock at release; however, their actuation time (time since the release command is given until release is effective) is longer and less predictable, whereas for pyrotechnic actuators the release is virtually immediate. Synchronised release is required for this mission, as all the connection points between mating parts have to be separated at the same time. Consequently, the pyrotechnic actuators have to be used for this mission.

9.1.3 Baseline Design

9.1.3.1 Spin and eject mechanism (SEM)

This mechanism is responsible for separation between the probe and the spacecraft. Connection between the probe and spacecraft is done at three nodes, spaced 120 degrees.

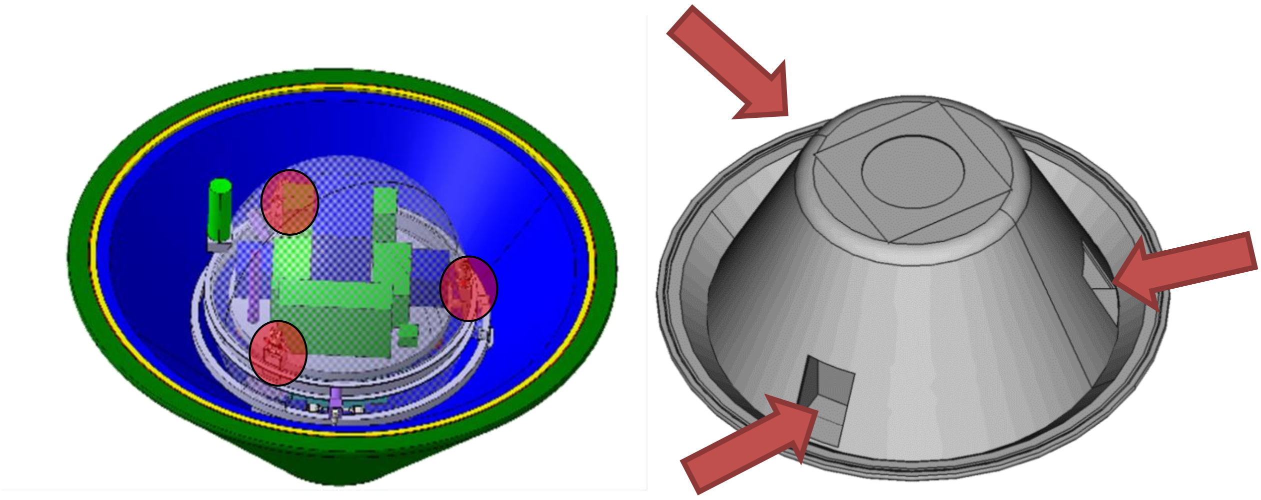

Figure 9-1: Location of mechanism nodes (red) on the probe (right) and corresponding apertures in the back shell, for the spacecraft-probe connecting rods (left)

The design of the mechanisms is based on the Huygens probe, of very similar configuration and mass properties.

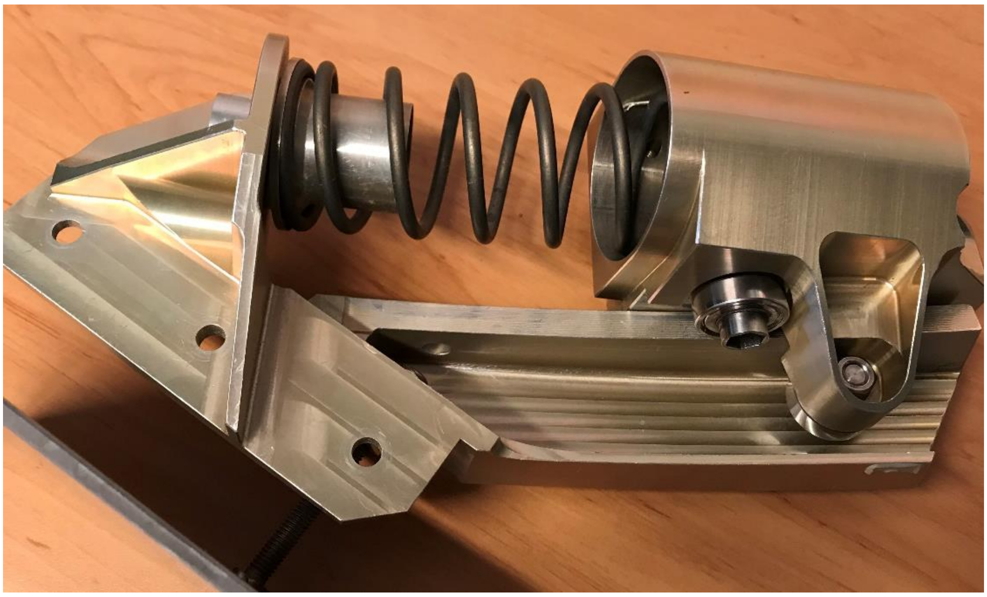

Figure 9-2: Spin and Eject Mechanism (1 out of the 3)

The part of the mechanisms attached to the probe is pushed by springs, and it follows the trajectory given by a helix-shaped guide. This provides the desired ratio between translation and rotation speed. Bearings are used to provide a smooth separation and to reduce risk of cold welding during transit.

Pyrotechnic release actuators initiate the separation of the probe in a synchronised manner. Separable connectors are considered as the baseline for the electrical connection with the spacecraft.

9.1.3.2 Front Shield separation mechanism (FSSM)

The Front Shield separation mechanism is also taken from the Huygens and PEP designs. Front shield and probe are connected at three equally spaced points. The location of these nodes is coincident with the Spin and Eject mechanism and the back cover separation mechanism. Pyrotechnic actuators are used to achieve a synchronised deployment and springs push the separating parts.

9.1.3.3 Back Shell separation mechanism (BSSM)

The Back Cover separation mechanism is also based on the Huygens and PEP designs. Back cover and probe are connected at the same nodes where the other mechanisms are located. Figure 9-3 shows the Front Shield and Back Cover separation mechanisms. Pyrotechnic actuators are used to achieve a synchronised deployment and springs push the separating parts away.

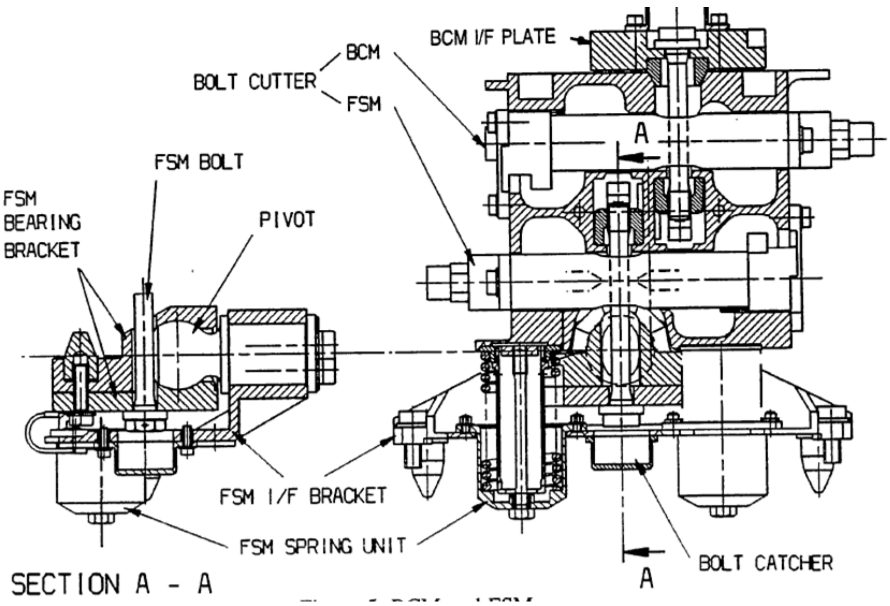

Figure 9-3: FSSM and BSSM (1 out of the 3 nodes) (RD[14])

9.1.3.4 Cable cutter for mortar-deployed pilot parachute

A cable cutter is also included, to cut the cable connection with the mortar-deployed pilot parachute.

9.1.3.5 Pyrotechnic actuators

Three pyrotechnic actuators are used per separation mechanism, one on each node. Two redundant European Standard Initiators are included in each pyro. This constitutes an internal cold redundancy concept (k=1 out of 2) at pyro actuator level.

Figure 9-4: Example of pyrotechnic actuators with 2 initiators

9.1.4 List of Equipment

In the computer model supporting this study, each mechanism is modelled as two different elements, corresponding to each of the separating parts for each mechanism.

9.1.4.1 Mass budget

Table 9-1 shows the mass budget coming from the mechanisms.

Mass budget			
Mechanism	Mass [kg]	Mass margin [%]	Mass with margin
SEM_SC	11.8	10	13
SEM_probe	2.4	10	2.6
FSSM_P	2.4	10	2.6
FSSM_DM	1.4	10	1.5
BSSM_P	0.9	10	1
BSSM_DM	1.4	10	1.5
Cable cutter	0.3	20	0.4

Table 9-1: Neptune Probe mechanisms mass budget

SEM_SC refers to the part of the mechanism that remains on the spacecraft, SEM_probe to the one on the probe. The suffix _P refers on mechanisms on the probe that do not stay on the final descent module. The suffix _D refers to the part of the mechanism staying on the descent module.

9.1.4.2 Power budget

The only power consuming elements are the pyrotechnic actuators.

Each pyro has a redundant firing circuit (cold internal redundancy k=1, n=2), power consumption of each one is:

E=0.15J

t=10ms max peak duration

P=E/t=15W average power

I=5A current

(Firing of 3 pyros should be included in the NASA spacecraft power budget for probe separation).

There are three pyros on each separation mechanism, plus one for the cable cutter for the parachute, totalling 10 pyros to be fired.

9.2 Orbiter

9.2.1 Requirements and Design Drivers

Deploy the magnetometer to a distance of 10m away from the spacecraft to achieve enough magnetic cleanliness at the magnetometer location, while providing enough stiffness. Distance to be deployed as well as stiffness required are to be detailed in later stages.

9.2.2 Assumptions and Trade-Offs

A deployable boom is required to separate the magnetometer from the spacecraft. This is the only mechanism identified for the orbiter itself, in addition to separation from the Solar Electric Propulsion (SEP) stage.

9.2.2.1 Trade off for the deployable boom

Several different concepts are available and proven for deployable booms, each with its advantages and disadvantages.

Coilable: Long length achievable, but lower stiffness and position accuracy. Besides, there is little European heritage and expertise regarding this type of boom.

Figure 9-5: coilable boom

(https://ttt.astro.su.se/groups/head/cost14/talks/Kallman.pdf)

Rigid articulated: high stiffness, heritage existing for the required length (10m), but higher mass and accommodation space required.

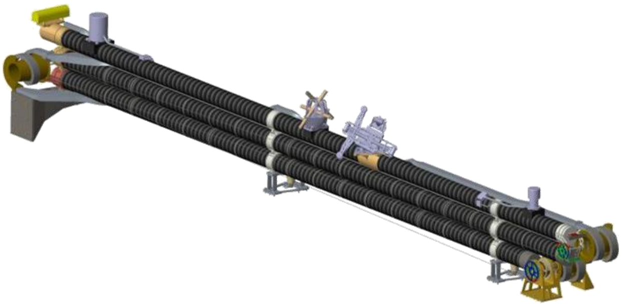

Figure 9-6: articulated boom

(www.aerospace.sener)

Collapsible: Low mass, but lower stiffness and risk of buckling for such a long boom.

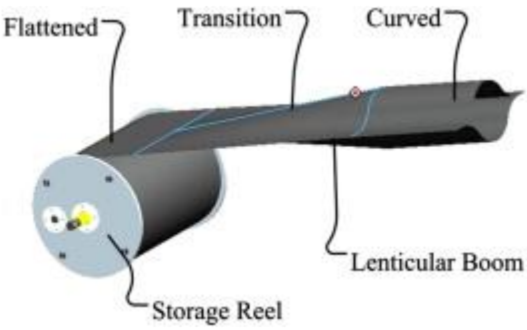

Figure 9-7: Collapsible boom (RD[16])

The choice for one technology is driven by the following criteria:

Risk is to be minimised, prioritising solutions with well-proven heritage and reliable design.

Mass is to be low enough not to become a driver of the mission, but will not be given priority above reliability.

Coupling with other disciplines is also considered. The solution has to be chosen in such a way that it minimises the risk of causing huge impacts in other disciplines in case the boom design has to be changed.

9.2.3 Baseline Design

The baseline design selected is an articulated rigid boom, based on the JUICE magnetometer boom. This boom is made of three CFRP tubes connected by rotary joints. The deployment is spring actuated. Three hold down and release mechanisms hold the boom in stowed configuration until deployment.

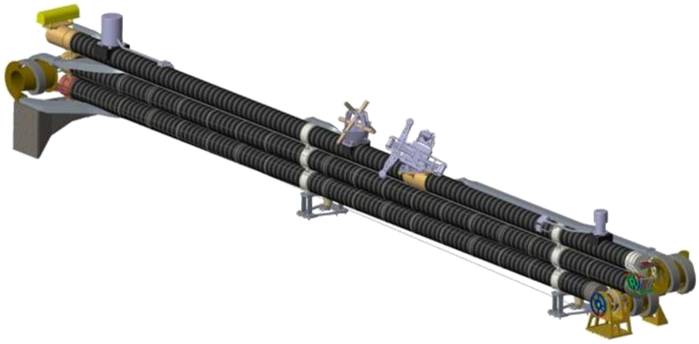

Figure 9-8: JUICE mag-boom

www.aerospace.sener

Due to the proposed orbiter being smaller than JUICE, the boom has to be folded in 4 segments instead of 3. Nonetheless, the JUICE technology can be adapted to this design with no major changes.

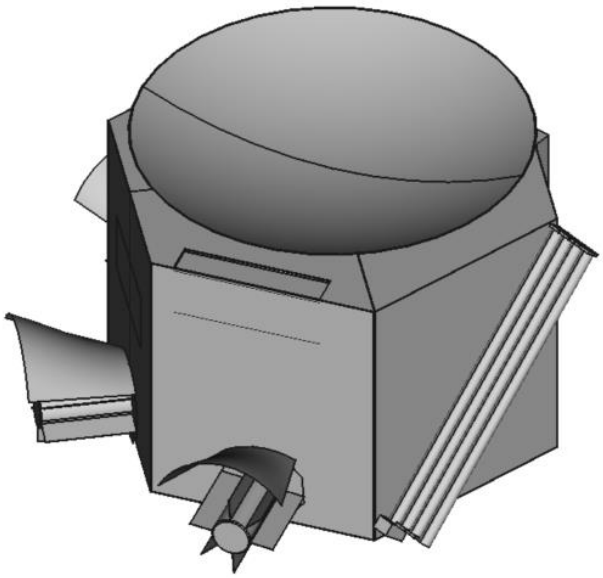

Figure 9-9: Boom accommodation in spacecraft

In later stages, once the requirements for the magnetometer are mature (distance to be deployed and positioning accuracy required), it is recommended to review the selection of boom technology, as other alternatives may prove more convenient. Coilable booms have been successfully used in several missions, and provide a much lighter solution in case a lower stiffness is acceptable.

9.2.4 List of Equipment

Mechanisms for the orbiter only include the magnetometer boom and the separation mechanisms from the SEP stage (only the portion of the mechanisms that remains on the spacecraft after separation). The following table shows the mass budget for the mechanisms.

Mass budget			
Mechanism	Mass [kg]	Mass margin [%]	Mass with margin
Magnetometer boom	30	10	33
SEP separation mechanisms	5	20	6

Table 9-2: Neptune Orbiter mechanisms mass budget

10 NEPTUNE PROPULSION

10.1 Orbiter

10.1.1 Requirements and Design Drivers

The requirements for the propulsion system are derived from the main requirements for the delta v and the mission and the ground operations.

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
PROP-010	Propulsion system provides necessary thrust and delta v for the mission manoeuvres	
PROP -020	Propulsion system provides torques to compensate the main thruster misalignments and for all other AOCS manoeuvres	
PROP -030	Propulsion system has at least three barriers for safety reasons on ground	
PROP -040	Propulsion system includes the measurement of the pressures within the subsystem at mandatory locations	
PROP -050	Propulsion system provides means to isolate potential mechanical pressure regulator leakage through the mission	
PROP -060	Propulsion system provides means to isolate the main engine in case of major leakage	
PROP -070	Propulsion system incorporates per branch a serial redundant pressure regulator	
PROP -080	Propulsion system includes Fill and drain valves for filling and testing of the propulsion system on ground	

10.1.2 Assumptions and Trade-Offs

The following table includes the assumptions used during the mission scenario.

Assumptions	
1	Gravity losses are linearly interpolated for the Apoposeidon radius [RN] of 275km, assumed to be representative for all mission cases. Additionally, a margin of 5% was taken for the delta v demands of the gravity losses.
2	The AOCS mass was modelled by using 5% of the total propellant mass used during the mission for the delta v manoeuvres. This propellant mass is split into 3% after the main Neptune Orbit insertion manoeuvre and 2 % after the mission.
3	For the bipropellant system, a mixture ratio of 1.65 was assumed. The dual mode systems used a mixture ratio of 1.43
4	No redundancy need for the main engine was assumed. It was furthermore assumed that the main engine can be accommodated in such a way to minimise the propellant need for any misalignment or centre of mass shift.
5	The thrust and the specific impulse of the engine was modelled by using the parametric model provided by the supplier.
6	The pressure set point in the tanks and the orifice of the main engine corresponds

Assumptions	
	to the nominal thrust point of 1kN for the engine, for the case of the mechanical pressure regulator. For the electronic pressure regulator case, adjustment to a thrust of 1.1kN is assumed to be possible.
7	The propellant mass includes a 2% residual mass of the propellant at the end of the mission.
8	The volumes of the tanks are calculated to fulfil the volume margin requirement of around 10%.
9	Neptune orbit insertion burn was done in one single manoeuvre. Tank depletion and corresponding temperature drop is assumed to be isentropic. Temperature threshold for the design and the helium tank size was around -18°C.
10	The tank sizes are derived from the E3000 tank family. Tank heights and masses are using linear interpolation.

Corresponding to assumption number 4, a discussion at IFP took place regarding the accommodation and possible shieldings needed. These shielding were thermal shields from engine to engine or shields in relation to micrometeroids. It was assumed at this stage that no special shielding is needed, either for thermal issues nor for micrometeroids. This has to be assessed in more detail for the final configuration of the Neptune orbiter as fixed at the Final presentation.

For all calculations, the following manoeuvre approach was used to estimate the propellant budget:

Manoeuvre	velocity increment [m/s]	propellant mass [kg]
Jupiter Flyby	15.00	
Neptune Insertion Manoeuvre	Delta v with gravity losses in comparison to thrust of propulsion system	
AOCS Mass		Propellant percentage of 3% for all delta v manoeuvres
Triton Target Manoeuvre	237.30	
Tour	68.25	
Tour	20.00	
Tour Margin	31.50	
Disposal	10.00	
AOCS mass		Propellant percentage of 2% for all delta v manoeuvres

Table 10-1: Delta V budget calculations

10.1.3 Baseline Design

The baseline design of the propulsion system consists of a pressurising section using helium and the propellant section using the propellant combination MON/MMH. It features two complete separated branches for the pressurisation of the propellant tanks, using serial redundant pressure regulator from VACCO (V1E10776-01). Both branches are isolated on ground using a nominal closed pyrovalve.

Downstream of the pressure regulators, one high pressure latch valve common for both branches and two check valves in each branch are accommodated to prevent propellant mixtures upstream the tanks and to provide the third barrier on ground. This has to be assessed in detail in a later stage of the mission including the assessment of possible liquid flowing towards the check valves during the entire mission scenario. Additionally, the benefits of having two latch valves including a possible entire isolation of a leaking branch by means of a dedicated latch valve should be traded against the disadvantages (mass, reliability) in a later stage of the mission analysis.

Downstream of the check valves, the four tanks are accommodated, two tanks for MMH and two tanks for MON. Due to the usage of the isovolumetric mixture ratio of 1.65 both tanks are similar in size and the depletion of the tanks will lead to a centre of mass shift only along one axis of the spacecraft.

The propellant section consists of two branches each has in common a latch valve and then separate equipment for the AOCS thrusters branches and the main engine branch. Since the baseline AOCS thruster includes a dual seat valve, only one latch valve to the main AOCS branch is sufficient. The main engine branches consist of one additional latch valve and Normally Open Pyrovalves to enable a potential isolation of the main engine. Since the failure of a huge leakage and the subsequent isolation of the main engine would lead maybe to a potential loss of mission, the need of this extra equipment should be traded as well in a later stage of the mission design.

Figure 10-1 shows the propulsion system in its designed stage. It only includes one line per propellant branch to the thruster to facilitate the reading of the schematic. In fact, every thruster has one connection to MMH and one to MON.

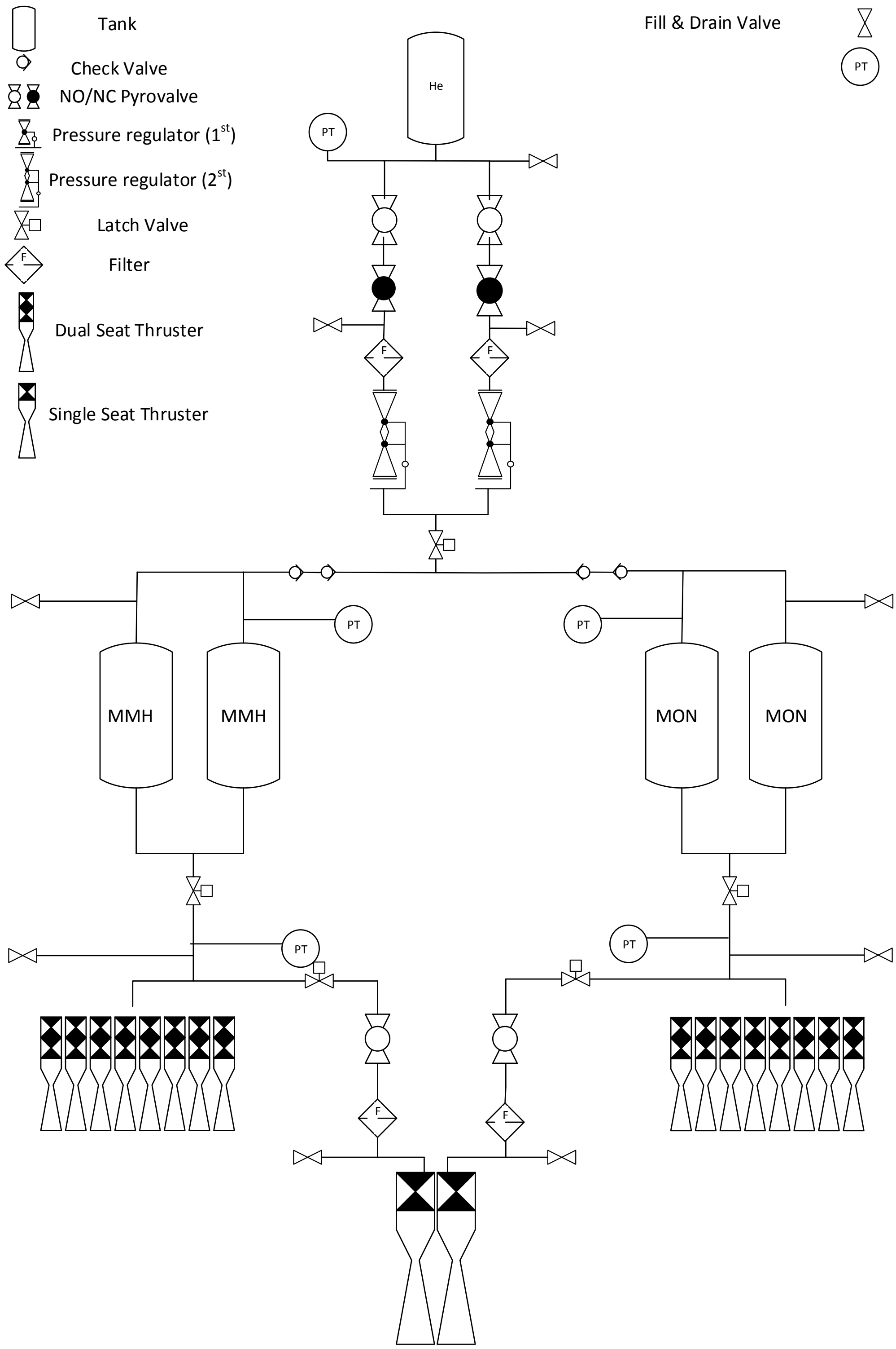

Figure 10-1: Propulsion system baseline schematic

The baseline design consists of two main engines RD[17] which has in its current stage planned to be trimmed to about 1kN of thrust. Due to the impact of the thrust ratio in relation to the mass as provided by the mission analysis, the system consisted of either only one engine or two engines of this kind.

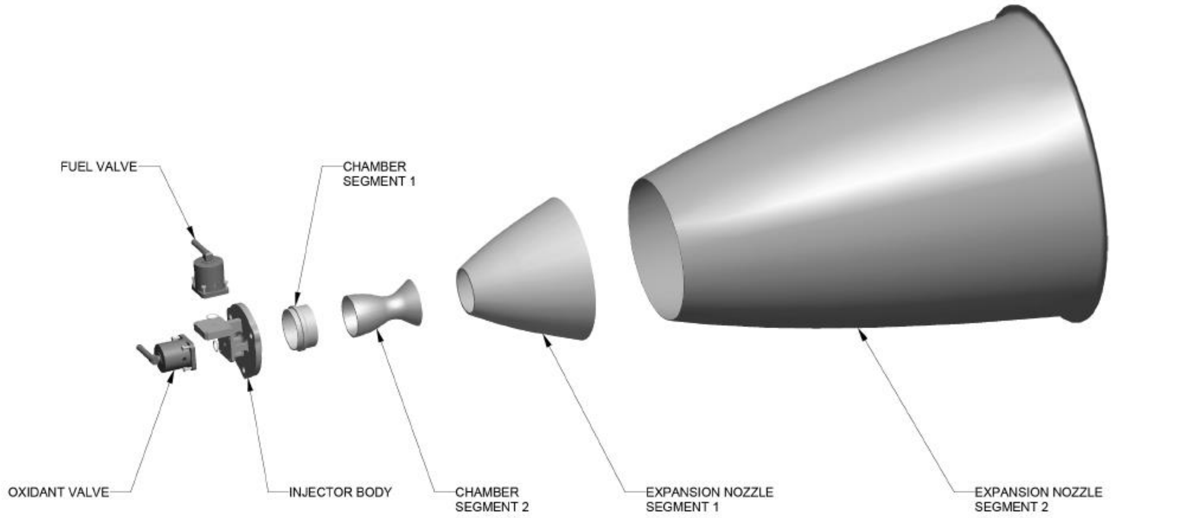

Figure 10-2: LEROS 4 (HTAE – High Thrust Apogee Engine) and its subassemblies

One of the main aspects highlighted for the configuration is the impact of thrust differences between the two main engines if the distance is quite high or the thermal radiation impact from one engine to the other if they are too close. Currently, the baseline was chosen to have two main engines but it has to be assessed in detail in a later study whether this accommodation and the drawbacks due to the thrust differences and the centre of mass shift is still the best possible solution. This should then also include a detailed assessment of the reliability of the system and the AOCS thruster as well as a detailed assessment of the thermal impacts from one engine to the other.

Furthermore, the system includes also AOCS thrusters, using the S10-18 thrusters manufactured by ArianeGroup RD[18].

Figure 10-3: S10-18 thruster from Arianegroup including the Plan for the thruster with main geometrical parameters

This thruster includes a dual seat valve to have two barriers in the main feeding line. The main characteristics of the thruster are:

10N Bipropellant Thruster Characteristics

Characteristics	Metric Values	Imperial/US Values
Thrust, Nominal	10N	2.2 lbf
Thrust Range	6.0 to 12.5N	1.4 - 2.8 lbf
Specific Impulse at Nominal Point	292 s	
Flow Rate, Nominal	3.50 g/s	
Flow Rate, Range	2.30 to 4.20 g/s	
Mixture Ratio, Nominal	1.60 to 1.65	
Mixture Ratio, Range	1.20 to 2.10	
Chamber Pressure, Nominal	9 bar	130 psi
Inlet Pressure Range	10 to 23 bar	145 - 335 psi
Throat Diameter (inner)	2.85 mm	0.11 inch
Nozzle End Diameter (inner)	35 mm	1.38 inch
Nozzle Expansion Ratio (by area)	150	
Mass Thruster with Single Seat Valve	350 g	0.8 lb
Mass Thruster with Dual Seat Valve	650 g	1.5 lb

Table 10-2: S10-18 Thruster characteristics

Chamber/Nozzle Material	Platinum / Rhodium alloy
Injector Type	Double Cone Vortex
Cooling Control	Film and Radiative
Propellants: - Fuel - Oxidiser	 MMH N2O4, MON-1, MON-3
Valve, Single Seat	Bipropellant torque motor valve
Valve, Dual Seat	Bipropellant torque or linear motor valve
Mounting Interface to Spacecraft	Valve flange with 3 feedthrough holes of 6.4 mm (¼") dia.
Tubing Interface	Per SAE AS4395E02, or welded
Valve Lead Wires	24 AWG per MIL-W-81381
Thruster Heater and Thermal Sensor	On request
Qualified longest single burn	8 Hours
Qualified Accumulated Burn Life	69 Hours
Qualified Cycle Life	1,000,000 cycles

Table 10-3: S10-18 Thruster characteristics (continued)

Using the LEROS 4 engine, the delta v demand for the Neptune Orbit insertion is around 3100s. Based on this, the temperature drop of the helium, assuming a starting temperature inside the tanks at the upper temperature level, the inlet temperature to the helium pressure regulator was estimated to be around -18.3°C which was assumed to be okay at this state. But this value has to be investigated in detail in a later stage.

The results of the system are the following:

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]
Jupiter Flyby	3968.59	3949.74	15.00	18.85
Neptune Insertion Manoeuvre	3949.74	1987.95	2162.55	1961.79
AOCS Mass	1987.95	1922.07	94.84	65.88
NOI clean-up	1922.07	1922.07	0.00	0.00
Triton Target Manoeuvre	1922.07	1782.59	237.30	139.48
PTTM clean up	1782.59	1782.59	0.00	0.00
Tour	1782.59	1744.38	68.25	38.21
Tour	1744.38	1732.03	20.00	12.35
Tour Margin	1732.03	1712.75	31.50	19.28
Disposal	1712.75	1706.68	10.00	6.07
AOCS mass	1706.68	1662.76	73.38	43.92
Final/Total (Including Residuals)	1604.60		2712.83	2351.96

Table 10-4: Baseline delta v and propellant mass results

Using the baseline design with only one engine (maybe an issue with the thermal radiation or the propellant mass needed for AOCS as mentioned above) would lead to an increase of the wet mass up to 4509.9kg. This is mainly due to the increase of the gravity losses for the Neptune orbit insertion manoeuvre. A detailed assessment of this manoeuvre and the misalignments/impacts of using two engines has to be done in the future. (A similar assessment was done, for example, for the ExoMars mission)

10.1.4 List of Equipment

The current baseline consists of the following equipment:

Description	Type	Amount	Mass per unit	Margin	Mass incl. margin
Pipes	Pipes	1	8	0.2	9.6
AOCS Engines	S10-18	16	0.65	0.05	10.92
Main Engine	LEROS-4	2	7.8	0.05	16.38
Fuel Tank	E3000	2	27.08	0.05	56.86
Oxid Tank	E3000	2	27.08	0.05	56.86
Fill / Drain Valves		9	0.07	0.05	0.6615

Description	Type	Amount	Mass per unit	Margin	Mass incl. margin
LP Pressure Transducer	SAPT	4	0.216	0.05	0.9072
HP Pressure Transducer	SAPT	1	0.216	0.05	0.2268
Latch Valve		4	0.75	0.05	3.15
Propellant Filter	RA04822A	4	0.077	0.05	0.3234
Check valve	VN005-001	4	0.585	0.05	2.457
Helium Tank	PVG-120	2	23.5	0.05	49.35
Helium Tank	PVG-65	1	12	0.05	12.6
Pressure regulator	VACCO	2	1	0.05	2.1
Pyrovalve	Cobham	4	0.315	0.05	1.323
SMA valve	Arianegroup	2	0.16	0.2	0.384
High pressure latch valve	Vacco V1E10560-01	1	0.8	0.05	0.84
Total					224.95

Table 10-5: Propulsion system (Neptune) Equipment list

Currently, two normally closed SMA valves as currently in qualification for MON/MMH systems are considered for the system RD[19]. In its current stage, those valves are only available in normally closed configuration but the supplier has indicated potential configurations as well for normally open designs. This could potentially be used as a replacement for the normally open pyro valves if there is any issue with radiation or lifetime limitation. However, this would come at the cost of qualification of such an equipment.

10.1.5 Options

Several options for the entire propulsion system were investigated. Corresponding list includes the investigations:

1. Dual Mode system using hydrazine/Mon and two engines.
2. Baseline design using an electronic pressure regulator with set point adjustments to have higher thrust level during the firing and using two engines.

Additionally, every concept was calculated by using only one engine.

The first one consists of using a dual mode system including two times the engine R-42DM from Aerojet RD[20] with a nominal thrust of 890N and the hydrazine thruster CHT-20 from ArianeGroup RD[21]. For the estimate, the propulsion system components except the tanks were kept constant, the tanks were estimated in the same family to be used with hydrazine as well. This system mass reduction was then used for the calculation with the impact of the higher gravity losses in comparison to the lower thrust level.

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]
Jupiter Flyby	3973.2	3954.7	15.0	19.03
Neptune Insertion Manoeuvre	3954.7	1989.3	2203.4	1969.84
AOCS Mass	1989.3	1923.0	69.8	66.19
NOI clean-up	1923.0	1923.0	0.0	0.00
Triton Target Manoeuvre	1923.0	1785.8	237.3	140.96
PTTM clean up	1785.8	1785.8	0.0	0.00
Tour	1785.8	1748.2	68.3	38.61
Tour	1748.2	1731.3	20.0	12.43
Tour Margin	1731.3	1705.0	31.5	19.40
Disposal	1705.0	1696.8	10	6.11
AOCS Mass	1696.8	1652.6	54.36049117	44.13
Final/Total (Including Residuals)	1594.1		2709.7	2363.03

Table 10-6: Dual Mode propulsion system results

The usage of only one engine would increase the delta v demand to 2579.36m/s. This would, using the percentage value for AOCS propellant mass as indicated above, lead to a wet mass increase up to 4562.2kg.

The second option was using the baseline design above but using an electronic pressure regulator to pressurise the tanks. This would enable the set point adjustment of the engines to reach a higher thrust. On the other hand, the system mass increase by having two electronic pressure regulators including the electronics are counterweighting the thrust benefit. Additionally, the thrust increase is leading to a slightly lower specific impulse which also affects the benefits of using the electronic pressure regulator.

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]
Jupiter Flyby	3990.5	3971.4	15.0	19.03
Neptune Insertion Manoeuvre	3971.4	2001.6	2150.2	1969.84
AOCS Mass	2001.6	1935.4	94.6	66.19
NOI clean-up	1935.4	1935.4	0.0	0.00
Triton Target Manoeuvre	1935.4	1794.5	237.3	140.96
PTTM clean up	1794.5	1794.5	0.0	0.00
Tour	1794.5	1755.9	68.3	38.61
Tour	1755.9	1743.4	20.0	12.43
Tour Margin	1743.4	1724.0	31.5	19.40
Disposal	1724.0	1717.9	10	6.11
AOCS mass	1717.9	1673.8	73.24	44.13
Final/Total (Including Residuals)	1615.42		2700.1	2363.03

Table 10-7: Baseline with Electronic Pressure regulator

Using this design but only one engine would increase the wet mass of the system to 4306.6kg. This wet mass, compared to the wet mass of the baseline with only one engine, is less because the delta v demands of the Neptune Orbit insertion manoeuvre is decreased. This is because the engine was assumed adjustable to about 1.1kN instead of only 1kN as currently planned for the nominal point. This benefit should be assessed in detail in the future.

10.1.6 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
	Electronic Pressure regulator	Pressure regulator using the high proportional valve for MON/MMH	Nammo (UK)	5		
	High Proportional Valve	This valve can be used for throttling purposes in MON/MMH/Hydrazine	Nammo (UK)	5		
	Normally Open Shape Memory Alloy valve	This valve could be used as a replacement for normally open pyrovalves without the lifetime limitation	ArianeGroup (Germany)	1-3		

11 NEPTUNE AOCS

11.1 Orbiter

11.1.1 Requirements and Design Drivers

		Subsystem Requirements		
ID	**Type**	**Statement**	**Parent ID**	**Comment**
AOGNC -010	F	The AOCS shall point the high-gain antenna to Earth for 3.2 hrs per (Earth) day and during safe mode or system checkout events		
AOGNC -020	M	The AOCS shall point the relevant instrument boresight to Neptune (anywhere from nadir to limb) during close approach period (100 hours per orbit with the exception of the 3.2 hours every 24 hours in which the spacecraft shall do the earth pointing for communication.)		
AOGNC -030	M	The AOCS shall point the relevant instrument boresight to Triton (anywhere from nadir to limb) during close approach period (22 hrs per orbit)		
AOGNC -040	M	The Absolute Performance Error (APE) of the instrument boresights shall be better than 300 arcsec half-cone angle at 95% confidence with temporal statistical interpretation		Based on JUICE / JANUS
AOGNC -050	M	The Relative Performance Error (RPE) of the instrument boresights shall be better than 1.5 arcsec half-cone angle over 100 msec at 95% confidence with temporal statistical interpretation		Based on JUICE / JANUS but assuming 10 times longer integration time due to low light
AOGNC -060	M	The Absolute Performance Error (APE) about the instrument boresights shall be better than 1 deg at 95% confidence with temporal statistical interpretation		Based on JUICE / JANUS
AOGNC -070	M	The Relative Performance Error (RPE) about the instrument boresights shall be better than 20 arcsec over 100 msec at 95% confidence with temporal statistical interpretation		Based on JUICE / JANUS but assuming 10 times longer integration time due to low light (10 times solar flux)

Subsystem Requirements				
ID	**Type**	**Statement**	**Parent ID**	**Comment**
AOGNC -080	D	The Absolute Performance Error (APE) of the high-gain antenna boresight shall be better than 1 deg half-cone angle at 95% confidence with temporal statistical interpretation		Derived from assumed antenna beam half-width of 1 deg
AOGNC -090	F	The AOCS should use different equipment for safe mode than is used in the normal mode		ESA best practice
AOGNC -100	M	The AOCS shall point the relevant instrument boresight to Jupiter (anywhere from nadir to limb) or Jovian moons during the Jupiter fly-by		Typical expectation for science mission passing a planet

11.1.2 Assumptions and Trade-Offs

Assumptions	
1	Systems checkout of the orbiter need only be performed at infrequent intervals (e.g. once per year) during cruise. Communications with orbiter is only mandatory during system checkouts.
2	Science camera cannot be used for navigation purposes (field of view and functional needs are not compatible)
3	Achieving adequate Neptune orbit injection burn accuracy requires IMU (accelerometer & gyro) for delta-v loop closure during burn
4	Laser communications is not considered as an option; note that it would drive pointing requirements if adopted
5	Mass moments of inertia are computed assuming the mass is isotropically distributed over a cylinder 2.5 m high with 2 m radius; relatively low height is necessary to reduce impact of main engine misalignments and the large radius improves the moment arm of RCS thrusters
6	If the orbiter must first insert into a low-periapsis (~2000 km altitude, needed for probe ejection) orbit, no science is required to be performed in this orbit. The periapsis is assumed to be raised (to ~20000 km) after ~10 days

11.1.2.1 AOCS reference frame

The frame assumed for AOCS design is illustrated below with the origin located at the geometric centre of the main body cylinder:

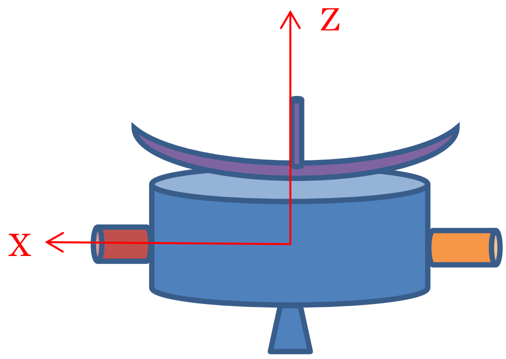

Figure 11-1: AOCS frame assumption; X – aligned with star tracker boresight, Z – aligned with high-gain antenna boresight. Orange cylinder represents payload camera and main engine thrust direction is +Z

11.1.2.2 Mass properties

Two mass states were studied:

- BOL – spacecraft with full wet mass – assumed during cruise phase
- EOL – end of fuel expenditure, dry mass only – assumed during science phase.

Using assumption 6 above, the diagonal mass moments of inertia are computed as:

- BOL: [5600, 5600, 7400] kg.m^2
- EOL: [2500, 2500, 3300] kg.m^2.

and the center of gravity in AOCS frame is assumed fixed at:

- [0.05, 0.05, -0.25] m (lateral offsets ~1% of s/c width).

Note that the minimisation of height in the design has the purpose of reducing the distance of the main engine to the c.g. (assumed to be 1 m), which drives misalignment torques. A large radius enables enhanced lever arm for RCS thrusters.

11.1.2.3 Orbit and fly-by trajectories

Several orbits and fly-by trajectories were considered for the Neptune orbiter mission:

1. Cruise: Heliocentric orbit with periapsis at Jupiter and apoapsis at Neptune
2. Post-probe-injection (if necessary): Low-periapsis Neptune orbit (2000 km altitude periapsis, 200 x $R_{neptune}$ apoapsis)
3. Science: High-periapsis Neptune orbit (20000 km altitude periapsis, 200 x $R_{neptune}$ apoapsis)
4. Science: Triton fly-by (during high-periapsis Neptune orbit) at 100 km altitude and relative speed of 3.9 km/s.

11.1.3 Baseline Design

The functional AOCS needs for the orbiter led to the selection of the following AOCS modes:

1. Acquisition & safe mode

2. Communications mode
3. Cruise mode
4. Science mode
5. Orbit control mode

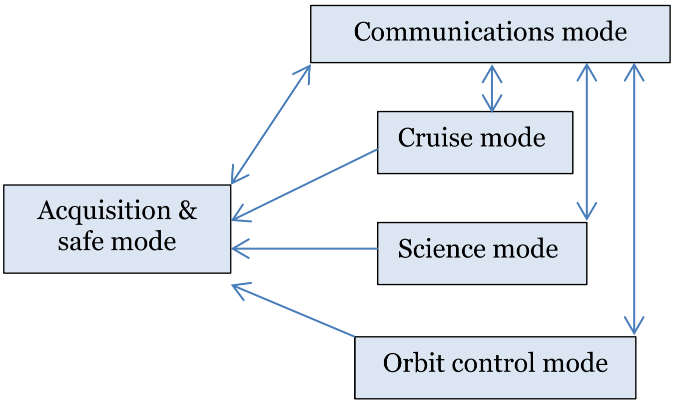

Figure 11-2: AOCS mode logic

In the above concept Communications mode is used as a bridging mode. The architectural design of each mode is described in section 11.1.3.1.

Note that a key lesson learned from Rosetta was that the number of modes and sub-modes should be limited to reduce testing burden. The different functional needs have been captured in this report thus a phase A should focus on consolidating mode design considering this desire for test efficiency.

The mapping from system modes to AOCS modes is as follows:

System Mode	AOCS Mode
LEOP	None/idle
Transfer mode	None/idle
Cruise mode	Acquisition & safe mode or Cruise mode
Manouevre mode	Orbit control mode
Observation mode	Science mode
Communication mode	Communications mode
Safe mode	Acquisition & safe mode

11.1.3.1 Mode designs

11.1.3.1.1 Acquisition & safe mode

This mode removes any residual spin rates from the spacecraft and acquires Earth-pointing for the high-gain antenna. There is no strong thermal constraints because the orbiter is not detached from the US carrier spacecraft until just before Jupiter fly-by, thus the solar heat energy is already low at this distance from the Sun. There are also no Sun-pointing constraints because the orbiter does not carry solar panels.

Previous outer-solar-system missions induced a search slew in safe mode to ensure that intermittently the antenna would sweep past the Earth. The known direction of the Sun, via a Sun sensor, could be used to decrease the search space to a cone using the known Sun-to-Earth offset angle from ephemerides. However, Earth-pointing can be more rapidly achieved by combining an Earth heliocentric position estimate, spacecraft position estimate and mapping between the inertial and body-reference frame via a star tracker. The subject of a star tracker based safe mode was studied in RD[22]. The star tracker would be loaded with a safe-mode set of parameters that are tuned for robustness rather than performance. This allows the star tracker to acquire at higher rates, for example. The required pointing accuracy for Earth-pointing is 1 deg (AOGNC-080), which is well within the capabilities of a low-performance (tuned for robustness) star tracker. The Earth and spacecraft ephemerides, propagated in the on-board software, would need to be loaded from ground and updated during the mission. Any reasonable-magnitude error in the spacecraft position estimate has negligible impact on the pointing accuracy given the > 500e6 km minimum distance from the spacecraft to Earth during autonomous operations.

The recovery from any spin motion can assume relatively low initial angular rates – a function of the ejection accuracy of the NASA carrier spacecraft. Subsequent entries to safe mode will also begin at low rates, depending on the FDIR threshold tuning for detection of RCS anomalies. An initial value of 2 deg/s has been assumed for propellant budgeting.

For cost and power reasons the recommendation is to embark a single star tracker in addition to a parallel-aligned cold redundant backup. A star tracker could be sufficient as sole sensor for this mode, but it may get blinded by the Sun or a nearby planet/moon.

One possible blinding-mitigation solution is to embark two additional optical heads and spread all the heads sufficiently far apart to ensure at least one non-blinded head at all times even in event of a single head failure.

Another blinding-mitigation solution is to execute an open loop thrust action to induce rotation orthogonal to the star tracker line of sight if it is blinded for a sufficiently long duration. However, this solution lacks robustness.

For robustness and cost reasons, the baseline solution is to add a coarse rate sensor (or an extra IMU/gyro of the same type used in orbit control mode), which also enhances the capabilities to handle higher than expected rates. This allows for closed-loop attitude control to stabilise rates and rotate the star tracker boresight (as needed) until it is no longer blinded. Note that since the star tracker is orthogonal to the high gain antenna there is no danger of Sun blinding after the Earth has been acquired. Planet or moon induced blindings are still possible, but on-board ephemerides of nearby planets or moons could be used to derive the attitude guidance quaternion such that the star tracker will always see cold sky (using degree of freedom of rotation about the high gain antenna boresight) or the coarse rate sensor could again be used to induce slow rotation (until blinding is removed) in a reactive manner.

The blinding-mitigation trade-off is presented in the table below:

	Coarse rate sensor	Extra IMU/gyro (same type as used in orbit manoeuvres)	Extra 2 star tracker optical heads
Total mass (including TRL margins)	0.9 kg	4.5 kg	5.9 kg
Total power (including TRL margins)	5.5 W	13.5 W (assuming gyro only; no accelerometer)	23.7 W (but only necessary to have all 4 heads on in safe mode)
Cost	Low	Medium	Medium

Table 11-1: Safe mode star tracker blinding mitigation trade-off

The safe mode attitude actuator should be different from that used in nominal modes – as per AOGNC-090. If nominal modes use wheels and RCS thrusters for momentum dumping then the safe mode should use a redundant RCS branch. If nominal mode uses thrusters at all times, then possibly a different type of thruster would be advised for use in the safe mode, or perhaps just the redundant RCS branch. Since wheels are the nominal modes baseline actuator, the safe mode will use a redundant RCS branch for actuation.

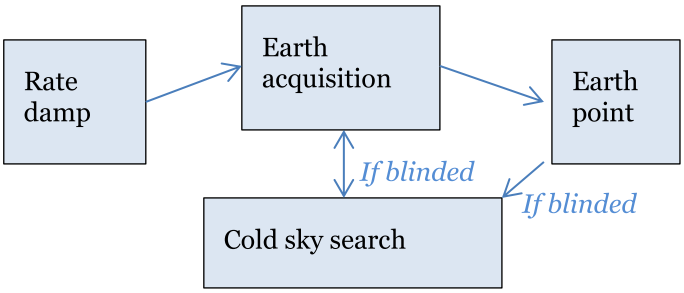

Figure 11-3: Safe mode logic

11.1.3.1.2 Communications mode

Earth-pointing for communications can be conducted in the same manner as for the safe mode, but with different equipment and more advanced handling of star tracker blindings.

Actuation can be performed with wheels for fine pointing and primary branch thrusters for momentum dumping and slews. This provides finer pointing and saves a small amount of fuel compared to using the thrusters alone (0.06 kg vs 0.02 kg), but thruster-only control is certainly an option if wheels are considered too power-hungry. 4 x reaction wheels will be embarked, but only 3 will be used at any one time in order to save power. Note that fuel estimates provided here are so low because the calculated limit cycle period is 12000 sec assuming 0.01 Ns MIB (minimum impulse bit) for a 10N RCS thruster and EoL inertias. Thruster pulsing is therefore very infrequent even when used as the primary actuator. However, fuel estimates are slightly optimistic because they assume steady state specific impulse.

Momentum dumping fuel is also very low because the environmental torque disturbances are relatively small in the nominal science orbit (orbit 3. Note that all disturbances are considered as secular because momentum dumps will be performed approximately daily to avoid oversizing wheels given the power constraints.

Attitude sensing can be done with the primary star tracker and attitude guidance can be planned (via on-board ephemerides) to avoid star tracker blinding from nearby planets or moons. If blinding is geometrically unavoidable, which is unlikely, one could consider temporarily operating the primary gyro (also necessary for closed-loop burns - see assumption 3) for propagating through anticipated outage periods. However, to minimise power consumption, which is especially important since the wheels draw a lot of power, it is preferred to leave the primary gyro off if not required. Pointing accuracy is sufficient with star-tracker-only given the communications APE requirement (AOGNC-080). If there are initial angular rates (e.g. from spin stabilisation during cruise) a sub-mode could be included to perform initial rate damping using the star tracker.

Although star trackers may be designed to handle solar flares, it may be difficult to entirely rule out unexpected outage events that affect both the nominal and the redundant tracker. This may be a problem as the gyros are not intended to be on all the time due to power restrictions. One could switch on the gyro temporarily in case of unsuccessful tracking on both optical heads. The gyro would then enable propagation from the last known attitude solution to rough accuracy - given that the bias would not have been estimated a priori by the on-board attitude estimator. Alternatively the attitude could just be left to drift during outages of both nominal and redundant star trackers since there is no strict requirements on thermal/power safe attitude range. If the power budget does allow for powering the gyro permanently in some spacecraft modes of operation it should be done. This was an important lesson learned on Rosetta, but Rosetta did not have the same power constraints as the Neptune mission.

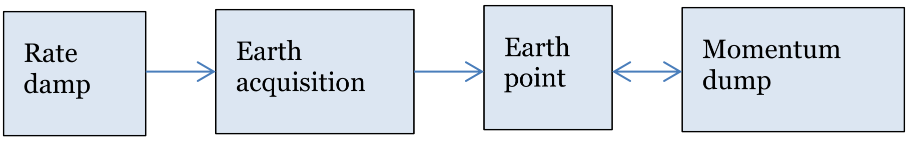

Figure 11-4: Communications mode logic

11.1.3.1.3 Science

Science requires pointing of instruments, such as a camera (JANUS) and visible/near-infrared spectrometeter (MAJIS) to planetary targets (Jupiter and Neptune) or moons (e.g. Triton). The specific target may be the limb, nadir or anywhere in between (AOGNC-020 & 030). ESOC requests that a navigation camera (and redundant backup) be included in the sensor suite to improve the estimates of spacecraft and moon ephemeris for targeting correction burns. The navigation camera would need to be pointed to planet or moon limbs during specific periods, possibly at the expense of science pointing depending on the relative layout of units on the orbiter.

It is intended to use a similar AOCS functional strategy to the Communications mode to achieve Science and navigation pointing objectives, albeit with different attitude guidance.

For periods in between science and communications, there are no constraints on attitude. During these periods the navigation camera could be pointed to the Neptune limb by default in order to maintain accurate spacecraft ephemeris. However, this decision is an open trade.

It is assumed that no rate damp sub-mode is required, providing one only transits to science mode from either Communications or Orbit control mode.

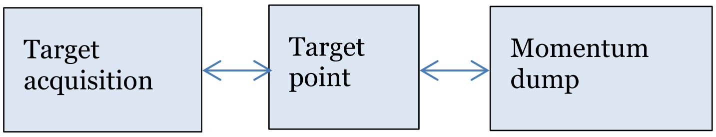

Figure 11-5: Science mode logic

11.1.3.1.4 Cruise

Several options have been considered for the AOCS strategy during the cruise:

1. Spin-stabilised s/c with continuous Earth-pointing of high gain antenna; requires < 1 deg nutation/coning and regular spin axis corrections to account for translational motion of spacecraft relative to Earth
2. Spin-stabilised to maintain at least an approximate known attitude, with temporary de-spin and transit to communications mode once per year for system checkout operations
3. 3-axis stabilised Earth pointing with wheels and momentum dumping with thrusters
4. 3-axis stabilised Earth pointing with thrusters.

Spin stabilisation with no more than 1 deg pointing error, assuming (for example) weekly spin-axis pointing corrections, requires a spin rate of ~0.02 deg/s given a maximum solar pressure disturbance of 6.6E-8 Nm. Since this is very slow spin and may be difficult to achieve accurately with thrusters, it is assumed instead an arbitrary nominal spin rate of 1 deg/s. The fuel required to generate the spin is 0.08 kg and fuel required to regularly re-orient the spin axis is 0.13 kg assuming a total slew angle of 180 deg over the 12 year cruise from Jupiter to Neptune.

Inertial spin-stabilisation with periods of interruption for communications requires 1.9 kg fuel for spin & de-spin (with 1 deg/s spin) and 0.01 kg fuel for slew and 3-axis pointing – assuming checkouts done one day per year. Spin-stabilisation options have the advantage that spacecraft equipment can be turned off during the majority of cruise to reduce the total operation hours of electronics or mechanisms (e.g. wheels). This could also be done without spin-stabilisation but then the attitude will be left to drift freely.

3-axis stabilised pointing with wheels requires that the wheels operate for a total of 14 years of the 16 year mission. This requires high reliability wheels, and this may not necessarily be possible given the necessity to select small-sat wheels to satisfy the power constraints (see section 11.1.3.4). The momentum dumping fuel consumption associated with this solution is negligible due to the weak solar pressure being the sole disturbance.

3-axis stabilised pointing with thrusters requires 0.35 kg fuel. Note that the fuel consumption is so small because the limit cycle period is computed to be 26000 secs during cruise phase. However, it still represents a large number of pulses over the lifetime, which could be an issue for qualification limits.

To avoid the need to run the wheels all the time during the cruise, and for a more robust Earth-pointing strategy for communications, the baseline design is to inertially spin-stabilise the satellite and turn the satellite equipment off during the cruise except for periods of system checkout – in which case the Communications mode is used. The annual running of the wheels will also help redistribute lubricant in the bearings to avoid the wheels getting stuck after a long period of non-use. Spin rate control should be done with the IMU as it is more accurate than the coarse rate sensor.

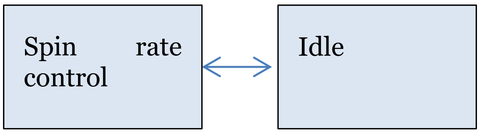

Figure 11-6: Cruise mode logic

11.1.3.1.5 Orbit Control

On basis of propulsion subsystem design trades, delta-V manoeuvres are executed using:

- Large delta-Vs: Single 1000 N engine with RCS in on-pulse mode for attitude control
- Small delta-Vs: RCS thrusters in off-pulse mode for attitude control.

To avoid switching off and on the wheels (which can affect lifetime) it is recommended to keep the wheels running at fixed speed during orbit control mode.

Attitude sensing can be performed by star tracker and gyro-based propagation in case of star tracker outage. The IMU will be running continuously during burns to keep track of the delta-V imparted and update the manoeuvre completion time.

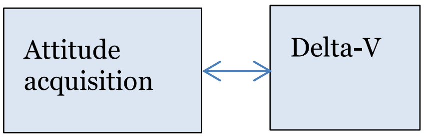

Figure 11-7: Orbit control mode logic

The option of spin-stabilisation during operation of the 1000 N engine should also be considered as it would allow the spacecraft shape to be narrower if desired and allow the star tracker to be switched off during large burns.

11.1.3.2 Equipment usage per mode

Mode	Acquisition & safe mode				Communications mode				Science mode			Cruise mode	Orbit control mode	
Sub-mode	Rate damp	Earth acquisition	Earth point	Cold sky search	Rate damp	Earth acquisition	Earth point	Momentum dump	Target acquisition	Target point	Momentum dump	Spin rate control	Attitude acquisition	Delta-V
Coarse Rate Sensor	X			X										
IMU												X		X
Star Tracker		X	X		X	X	X		X	X			X	X
Navigation Camera										(X)				
Reaction Wheels					(X)	(X)	X	X	(X)	X	X		(X)	(X)
RCS	X	X	X	X	X	X		X	X		X	X	X	X

Table 11-2: AOCS equipment usage per mode and sub-mode

Note in the above table that (X) means the unit is used in a secondary manner. For wheels (X) implies fixed-speed control of reaction wheels to avoid having to turn them off and on. For the navigation camera (X) means that it is only used during periods where ground wishes to improve knowledge of planetary or moon ephemerides.

11.1.3.3 Sensor selection

11.1.3.3.1 Star tracker

Since the TID for the Neptune mission is estimated at ~66 krad, several high performance European star trackers would make suitable candidates. However, since there is also a requirement to perform accurate science pointing around Jupiter (AOGNC-100) on the way to Neptune, it is logical to embark the JUICE-version of the Sodern Hydra. This unit has been designed and shielded specifically for being able to acquire and track stars in the high radiation environment around Jupiter.

The baseline design includes the following hardware in cold redundancy:

- 2 x Sodern JUICE-Hydra optical heads (see RD[23])
- 2 x cross-strapped Sodern JUICE-Hydra electronic units.

As mentioned previously, since the choice is to operate just a single star tracker (to save power) it should be mounted parallel but opposite to the payload boresight since the tracker has the best accuracy orthogonal to its boresight – which is also the most important axes for the payload.

If there has already been a star tracker failure previously and thus the spacecraft is already operating on the redundant optical head or electronic unit, then the baseline design is simply to continue using the redundant unit if a safe mode transition is initiated. Failure of both nominal and redundant unit could be considered a double failure thus out of scope. However, if project policy is not to rely on the exact same hardware in normal mode and safe mode then an extra electronics unit and optical head could be embarked for exclusive use in the star-tracker based safe mode.

11.1.3.3.2 Navigation camera

NASA typically uses the payload camera for improving knowledge of spacecraft or moon locations with respect to a nearby planet. However, ESA prefers functional separation of the navigation task from the science task. Furthermore the baseline science camera has a very narrow field of view thus is less suited to the task of limb fitting. For these reasons the baseline design includes the following hardware in cold redundancy:

- 2 x Sodern JUICE navigation camera with electronics (see RD[24]).

The navigation camera should be approximately aligned with the science payload to reduce the need for separate pointing sessions for navigation updates. The camera need only be operated at intermittent intervals as required by ground for targeting of delta-V burns. In nominal operations the camera is capable of producing a relative planet/moon position estimate every 15 minutes. The images can also be relayed to the ground during communications mode for solution cross-comparison by the mission operations centre.

11.1.3.3.3 IMU

Several options are available for long lifetime / high reliability IMUs. The European Astrix 1090A includes a medium performance fibre optic gyro and a Honeywell QA3000 accelerometer. It has been embarked on ExoMars and is considered suitable for a mission to Neptune, with a radiation hardness of 100 krad. Note that the gyro is only used for spin-rate control during cruise and for decoupling translation and rotation motion during the delta-V burns, thus high performance is not needed; the star tracker is used as primary attitude sensor during burns.

For a ~1.5 hour burn to execute the 2161 m/s Neptune injection delta-V, the 1090A expected delta-V estimation error is expected to be < 10 m/s.

The baseline design includes the following hardware in cold redundancy:

- 2 x Airbus Defence & Space Astrix 1090A IMUs.

11.1.3.3.4 Coarse rate sensor

The safe mode design description identifies a need for a coarse rate sensor to aid a cold sky search in the event of star tracker blinding. The performance requirements for such a sensor are very loose. Several low cost options are available, but the best option is considered to be the SiREUS NG10 since they use radiation hardened parts and do not have obsolescence issues.

The baseline design includes the following hardware for use in safe mode only (without redundancy):

- 1 x TAS SiREUS NG10.

The specific radiation susceptibility level information was not available for this study, but it is possible that some additional shielding may be required.

11.1.3.4 Actuator selection

11.1.3.4.1 Environmental torque disturbances

Simulations were conducted using ESA's GAST tool, setup for orbit about Neptune. Relevant assumptions were:

- Residual magnetic moment: 1.6 Am^2, based on mass-based formula from NASA SP8018 standard
- Offset of Z-axis from nadir during Earth communications: 45°

- s/c solar pressure reflectance factor: 0.5
- Solar pressure cross-sectional area: 12.5 m².
- Neptune magnetic field dipole as per RD[25].

Simulations of length 1 Neptune day were conducted near periapsis and apoapsis and these were used to check the static analytical torque estimates from the AOGNC Excel workbook tool. The simulation results agreed with the workbook to within an order of magnitude. The workbook was then used to find the mean torque disturbances over the orbit which is useful for propellant budgeting. The mean value is found numerically by computing torques at each 10 deg increment of mean anomaly and then averaging these values.

The simulation results below are given for the temporary low-periapsis orbit (orbit 2) used just after injection, which is required in the case that the orbiter carries an atmospheric probe. Figure 11-8 shows that the orbiter is only close to Neptune (within 1 $R_{neptune}$ altitude) for around 10000 secs, where it sweeps through a ~180 deg change in true anomaly.

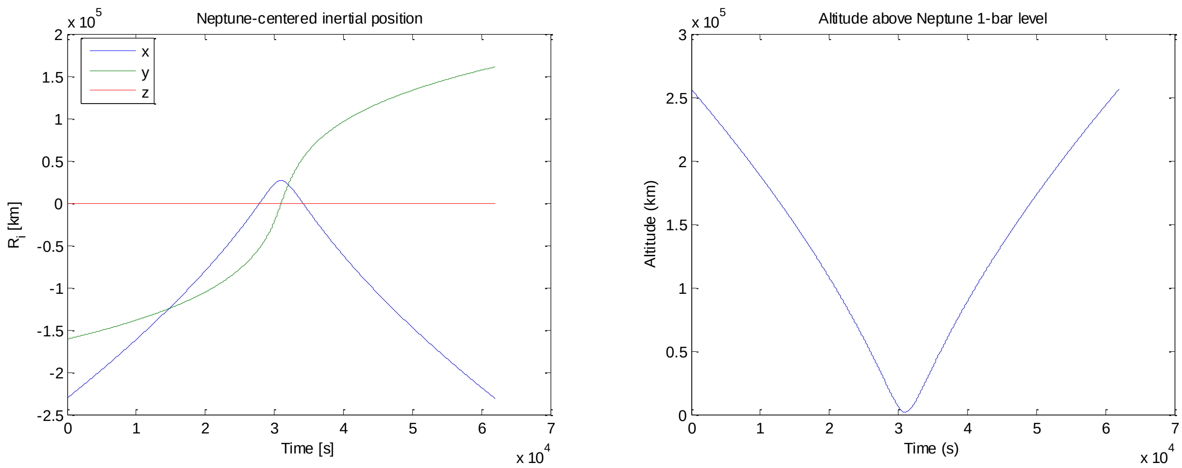

Figure 11-8: Low-periapsis orbit, periapsis pass, inertial pointing. Position in Neptune-centred inertial frame (left) and altitude (right)

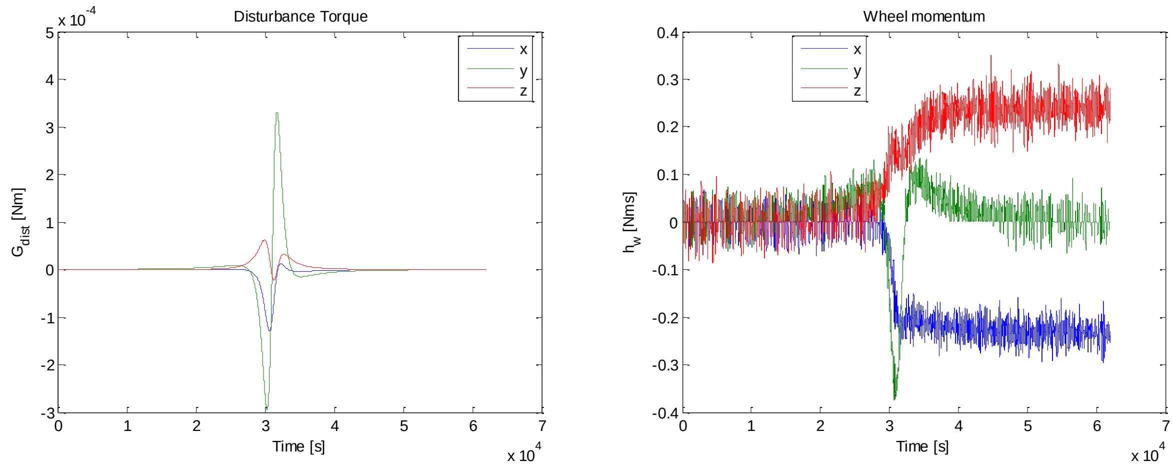

Figure 11-9: Low-periapsis orbit, periapsis pass, inertial pointing. Disturbance torques (left) and wheel momenta (right)

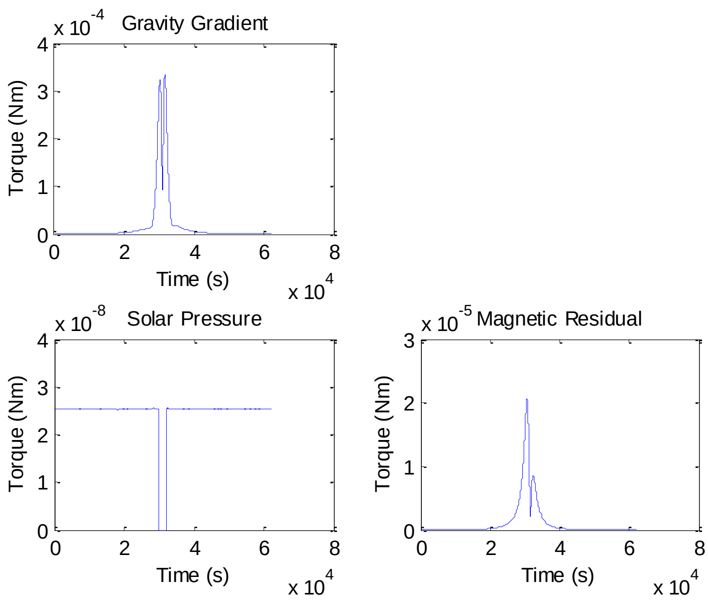

Figure 11-10: Low-periapsis orbit, periapsis pass, inertial pointing. Disturbance torque contributors

Figure 11-9 and Figure 11-10 show that the gravity gradient disturbance torque is dominating, due to off-pointing from nadir and the strong gravity of Neptune. However, the gravity gradient torque profile after periapsis pass is equal and opposite (sign) to that prior to periapsis pass thus the momentum accumulation is temporary. The secular accumulation is just 0.3 Nms per axis. It is assumed in the simulation that 3 wheels are mounted in orthogonal configuration for simplicity. In the above plots the angle of body +Z w.r.t. nadir is 45 deg (gravity gradient worst case) at t=0 and then changes rapidly to ~150 deg at periapsis pass. The worst case would be a nadir offset of 45 or 135 deg at periapsis pass, however a repeated simulation with this constraint showed that torque and momentum accumulation is still roughly the same as that shown above.

The torques at apogee (Figure 11-11) are very small and dominated by solar pressure, which is of course present throughout the entire mission.

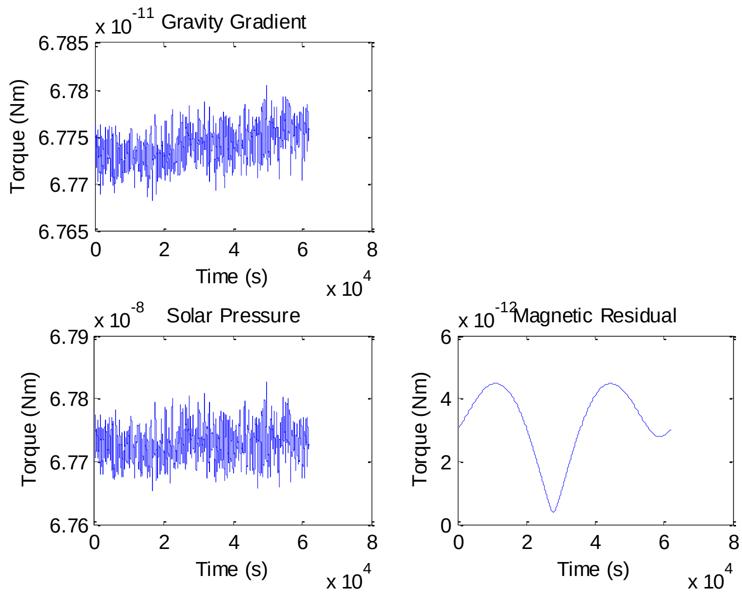

Figure 11-11: Low-periapsis orbit, apoapsis pass, inertial pointing. Disturbance torque contributors

After ~10 days, the periapsis will be raised to ~200000 km – the science orbit. The results below show the momentum impact of reduced disturbance torques for the new orbit (orbit 3).

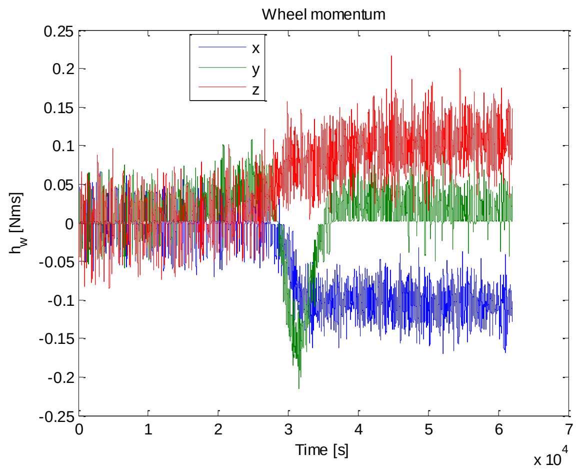

Figure 11-12: Science orbit, periapsis pass, inertial pointing. Momentum accumulation

In the science orbit, the orbiter will be switched between inertial/Earth pointing (as above) for downloading data and orbit-frame-fixed science attitude for gathering science data. Although it is possible it may be needed to point science and navigation cameras to the limb of Neptune, the majority of the time will presumably be spent in nadir-pointing attitude. This has been simulated and the results are presented below.

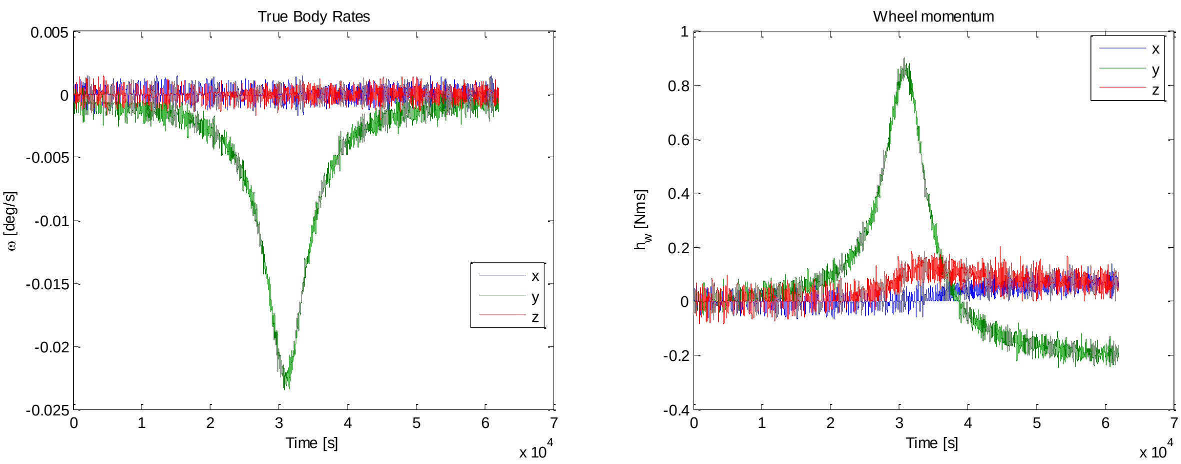

Figure 11-13: Science orbit, periapsis pass, nadir pointing. Body-frame angular rates (left) and wheel momenta (right)

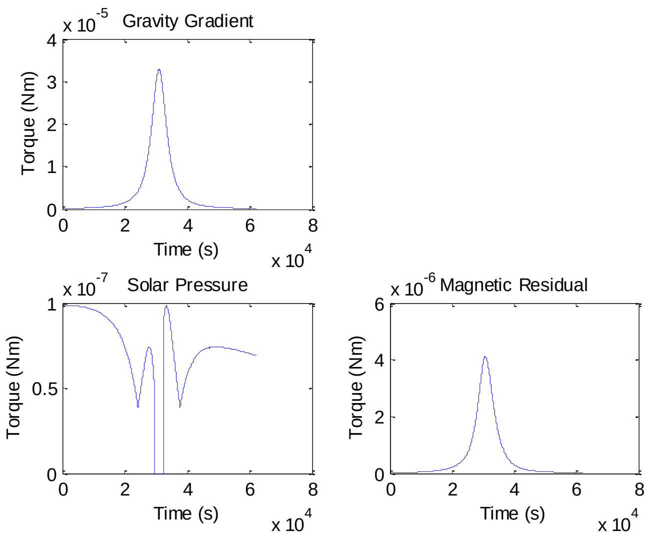

Figure 11-14: Science orbit, periapsis pass, nadir pointing. Disturbance torque contributors

The momentum accumulation in nadir pointing attitude is similar in magnitude but most pronounced on the Y axis instead of the X and Z axes as was the case with inertial pointing. There is also a large momentum transient due to the fast rate of change of the orbital frame around periapsis (see Figure 11-13). This is a consequence of the highly eccentric orbit geometry.

The disturbances are summarised in the table below:

	Neptune Pointing			**Earth Pointing**			**Earth Pointing (with low pericenter)**		
	Pericenter	*Apocenter*	*Orbital mean*	*Pericenter*	*Apocenter*	*Orbital mean*	*Pericenter*	*Apocenter*	*Orbital mean*
Torques (Nm)	Max from simulation over 1 Neptune rotation period		Integrated using AOGNC workbook	Max from simulation over 1 Neptune rotation period		Integrated using AOGNC workbook	Max from simulation over 1 Neptune rotation period		Integrated using AOGNC workbook
Magnetic	4.E-06	4.E-12	2.E-07	4.E-06	4.E-12	2.E-07	2.E-05	4.E-12	1.E-06
Gravity Gradien	3.E-05	2.E-11	2.E-06	1.E-04	7.E-11	4.E-06	3.E-04	7.E-11	2.E-05
Solar Pressure	1.E-07	3.E-08	2.E-08	4.E-08	7.E-08	2.E-08	3.E-08	7.E-08	2.E-08
Aerodynamic	negligible	negligible	negligible	negligible	negligible	negligible	negligible	negligible	negligible
Total Momentum Accumulation (Nms / Neptune-day)	0.2	small	0.1	0.1	small	0.3	0.3	small	1.3
Momentum Max Transient (Nms)	0.9	-> due to rate guidance		0.2	-> due to gravity gradient		0.4	-> due to gravity gradient	

Table 11-3: Environmental torque disturbances summary

The orbital mean torque for gravity gradient seems to be over estimated because the workbook assumes a constant offset from nadir, rather than time varying, and because it does not account for the fact that the high gravity gradient during the half-day prior to periapsis pass is mostly balanced by the gravity gradient the half-day after periapsis pass; this period of time dominates the orbital mean. Therefore, for sizing momentum devices the simulation values for pericenter pass will be used rather than the orbital mean values quoted from the AOGNC workbook.

11.1.3.4.2 Fine pointing actuator

Fine pointing could either be done with:

- Reaction wheels, and regular momentum dumping with RCS thrusters
- Cold gas thrusters
- RCS thrusters.

A trade-off is provided below:

	Reaction wheels	Cold gas	RCS
Total mass (including TRL margins)	12.4 kg, assuming Honeywell HR04 wheels (see sizing in next section)	18 kg, assuming 8 thrusters and scaled-down Euclid system	10 N thrusters: no extra dry mass since equipment already embarked for other reasons 1 N thrusters: dry mass 3 kg (assuming 8 thrusters) + piping Extra fuel: 2.2 kg including 100% margin (but this fits in ample margins taken for fuel budget)
Total power (including TRL margins)	29 W (assuming mean operation at 30% of momentum capacity and including a 20% TRL margin)	23 W base + 16 W if all thrusters on, but duty cycle will be low and base power could be reduced if throttling controller removed	17 W if all thrusters on but duty cycle will be very low, with exception of periapsis pass
Cost	Medium	High	Low
Pointing accuracy	High	Very high	Medium; not certain if RPE requirements can be met
Other comments	-May require some redesign to include radiation hardened parts or other modifications to guarantee sufficient lifetime.	-Complex architecture given simultaneous use of Bi-propellant propulsion system -Production line may be discontinued	-Could embark additional mono-propellant 1N thrusters to further reduce MIB

Table 11-4: Fine pointing actuator trade-off (green indicates baseline)

A cold gas system was quickly eliminated from the trade due to its complexity and questions over the production line.

Given the relatively stringent RPE requirements, it is not certain whether RCS-based control would be sufficiently smooth especially around periapsis pass where torque adjustments will be frequent. In the majority of the orbit the disturbances and required angular rates are very low and RCS-based control will most likely suffice. However, if orbit-frame-fixed pointing needs to be maintained during periapsis pass the microvibrations from frequent RCS firings may violate the RPE requirements and result in blurry imagery.

Since reaction wheels offer better pointing performance than RCS control and because the power budget can accommodate them (albeit without margin to grow), wheels are included in the baseline. However, this is a crucial first point of iteration should this mission design move forward to a Phase A. NASA's New Horizons probe did all fine pointing with 1 N RCS thrusters thus it is expected that a no-reaction-wheel design is possible.

11.1.3.4.3 Reaction wheel sizing

From Table 11-3, the worst case science mode needs for the reaction wheels (taking the simulation results in favour of the AOGNC workbook estimates) is the 0.9 Nms transient momentum peak required to maintain orbit-frame-fixed pointing during periapsis pass.

For slews between science attitude and Earth communications attitude, a 180 deg 1 hour (reasonable from availability standpoint) slew would require 8 Nms wheels. Wheels of this size consume ~20 W in steady state per wheel, i.e. 60 W total.

For the closest-approach Triton fly-by (see section 11.1.2.3) nadir-pointing would require 6 Nms wheels.

Given the extreme power constraints on this mission, a 1 Nms wheel is selected as baseline. It is assumed that slews can be performed completely with thrusters and that fly-bys will either have to tolerate increased absolute pointing error due to limited slew rate from 1 Nms wheels or the fly-by can be thruster-assisted. No special allowance is made for this in the AOCS mode structure yet but it is something that should be considered in a phase A design.

Since ESA interplanetary missions always demand use of high reliability parts, the typical suppliers of 1 Nms wheels (for the smallsat/microsat market) cannot be considered. Honeywell supply the HR04 1 Nms wheel, which is a possible candidate – consuming ~8 W steady state per wheel. However, this is advertised as just 5 year minimum mission life thus would possibly require some parts replacement to qualify it for Ice Giants (16 years, including 2+ years of wheel operations).

The baseline design includes the following hardware with 3 active wheels and 1 in cold redundancy:

- 4 x Honeywell HR04 reaction wheels, in pyramid configuration.

Rockwell Collins Deutschland is a candidate European supplier that could be considered. There is evidence that they do supply wheels in this class but do not seem to actively advertise them. Astrofein or MSCI also supply wheels of this size but their use of COTS parts would need to be thoroughly revised to meet reliability/radiation/lifetime requirements for Ice Giants.

11.1.3.4.4 Thruster sizing

RCS thruster size is driven by the need to provide torques to counteract the misalignment torques induced by the main engine.

Assuming 1 deg misalignment of the 1000 N engine, placed 1 m from the c.g., with RCS thrusters at a moment arm of 2 m, the RCS thrusters must be at least 6.5 N with a 4-thruster box configuration and 15 deg tilt to enable Z-axis control. 10 N thrusters are available in bi-prop configuration therefore these are selected for Ice Giants.

These results are obtained assuming a 20kg mass, assuming different configurations and a 5% error margin, the mass can becomes 110 kg and the requirement in Newton for the thrusters stay the same.

Assuming a thruster with a Minimum Impulse Bit of 0.01Ns and a moment arm of 2 meter, the speed induced on the Z axis with dry mass is 0.02deg/min which is considered small.

If two main engines are embarked and will be fired simultaneously then larger RCS thrusters should be embarked or spin-stabilisation during main engine firings should be employed.

The number of thrusters required depends on the level of fuel-use, efficiency desired and any need for torque control to have no impact on the orbit. Since neither of these have been expressed as strong needs for Ice Giants a simple 4 thruster configuration should suffice. However, the propulsion design baselines 2 strings of 8 thrusters per string in case force-free torques become a requirement. These could be laid out as two opposing box configurations with tilted thrusters or optimisation could be performed.

11.1.3.5 Attitude control propellant budget

The attitude control propellant uses the following input data:

- Phase duration
  - Science phase: 66% of 2 years
  - Communications phase: 33% of 2 years
- Environmental disturbances
  - Mean total torques over orbit as per Table 11-3
- RCS properties
  - 4 x 10 N thrusters, with 15 deg tilt to achieve Z-axis control
  - MIB 0.01 Ns
  - Specific impulse 290 sec
- Delta-V firings
  - 2618 m/s total main engine usage
  - Engine misalignment and RCS-layout assumptions as per previous sub-section
  - Control overshoot margin: 10%
- Cruise phase
  - Spin up/down during cruise: from ejection with initial rate 2 deg/s and then 24 repeats for rate change of 1 deg/s spin stabilisation between system checkouts
  - Slews cruise phase: 180 x 1 deg, 1 hour slews (for system check out & comms)
  - Fly-by pointing (i.e. Jupiter and Jovian moons) during cruise phase: 12 fly-bys of 1000 km closest approach at 5 km/s relative speed
- Science phase
  - Slews between communications and science attitude: 180 deg, twice per day over two years

  - Fly-by pointing (i.e. Triton, etc.) during science phase: once per month over two years with 1500 km closest approach (to moon centre) at 3.9 km/s relative speed
- Safe mode
  - 3 de-spins (from 2 deg/s), 3 x 180 deg, 1 hour slews and 10 days Earth-pointing with 1 deg deadband.

All values in kg	Mass	w/ 100% margin
Attitude control during main engine firings	14.4	28.7
Control overshoot margin for main engine firings	1.4	2.9
Slews transfer phase	0.1	0.1
Slews science phase	1.9	3.8
Planet/moon fly-by rate assist, transfer phase	0.3	0.6
Moon fly-by rate assist pointing, science phase	0.1	0.1
Spin-stabilize/recovery	1.9	3.8
3-axis stabilized safe mode	0.1	0.2
Wheel-momentum dump Neptune-nadir attitude	0.1	0.1
Wheel-momentum dump communications or moon-pointing attitude	0.1	0.1
TOTAL		40.4

Table 11-5: Attitude control propellant budget

Note that the propulsion subsystem has allocated much more than this (roughly double) for attitude control propellant to remain conservative and to help offset the optimistic assumption of steady state specific impulse used to compute the above budget. Also note that stand alone fuel values reported earlier in this chapter do not contain any margin unless explicitly stated.

As mentioned in the tradeoff for the fine pointing actuator, if the reaction wheels were removed and all fine pointing was done with thrusters an additional ~2.2 kg of propellant would be required including 100% margin.

11.1.3.6 Pointing budgets

The primary contributors to the camera payload pointing APE and RPE budgets are given in the tables below.

Payload pointing error	About payload LoS	Transverse to payload LoS
Post-calibration payload alignment knowledge error (estimate after discussion with ESA JUICE GNC lead)	160	16
Attitude guidance error bias (along-track nav. error of 3 km assumed)	negligible	46 (Neptune) or 920 (Triton)

Payload pointing error	About payload LoS	Transverse to payload LoS
Star tracker bias (Hydra)	12	8
Star tracker noise equivalent angle (Hydra with transfer function)	4.4	0.6
Rate estimation error (Hydra with transfer function)	6.1	0.8
Controller delay (1 x 8 Hz cycle)	1.6	negligible
Magnetetomer boom flexible oscillations (neglecting flex filtering in controller design)	6.6	0.4
TOTAL *(RSS summation within error categories and linear summation of categories)*	174	65 (Neptune) or 940 (Triton)

Table 11-6: Science cameras' pointing APE; all values are arcsec and are given at 2-sigma confidence

Payload pointing error	About payload LoS	Transverse to payload LoS
Star tracker noise equivalent angle (Hydra with transfer function)	4.4	0.6
Rate estimation error (Hydra with transfer function)	6.1	0.8
Controller delay (1 x 8 Hz cycle)	1.6	negligible
Magnetetomer boom flexible oscillations (neglecting flex filtering in controller design)	6.6	0.4
TOTAL *(RSS summation within error categories and linear summation of categories)*	13	1.2

Table 11-7: Science cameras' pointing RPE; all values are arcsec and are given at 2-sigma confidence

The above estimated performances satisfy all the APE and RPE requirements (AOGNC-040 to -070) with the exception of APE during Triton pointing – dominated by navigation error due to the small moon size and close approach. It is likely that the error can be better than the 3 km value estimated here.

The APE budget for communications mode is not presented, though it will be dominated by post-calibration antenna residual misalignment bias error. It is expected to be compliant.

11.1.4 List of Equipment

	mass (kg)	mass margin (%)	mass incl. margin (kg)
SC (Spacecraft)	56.00	7.86	60.40
GYRO_Sireus (GYRO Selex Galileo Sireus)	0.80	10.00	0.88
IMU_Astrix_1090A_1 (IMU Airbus Astrix 1090A #1)	5.00	5.00	5.25
IMU_Astrix_1090A_2 (IMU Airbus Astrix 1090A #2)	5.00	5.00	5.25
NavCam_1 (NavCam #1)	11.00	5.00	11.55
NavCam_2 (NavCam #2)	11.00	5.00	11.55
RW_HR04_1 (RW Honeywell HR04 #1)	2.60	20.00	3.12
RW_HR04_2 (RW Honeywell HR04 #2)	2.60	20.00	3.12
RW_HR04_3 (RW Honeywell HR04 #3)	2.60	20.00	3.12
RW_HR04_4 (RW Honeywell HR04 #4)	2.60	20.00	3.12
STR_HydraEU_Juice_1 (STR Sodern Hydra JUICE Electronics Unit #1)	3.60	5.00	3.78
STR_HydraEU_Juice_2 (STR Sodern Hydra JUICE Electronics Unit #2)	3.60	5.00	3.78
STR_HydraOH_Juice_1 (STR Sodern Hydra JUICE Optical Head #1)	2.80	5.00	2.94
STR_HydraOH_Juice_2 (STR Sodern Hydra JUICE Optical Head #2)	2.80	5.00	2.94
Grand Total	56.00	7.86	60.40

Power (W)		
	P_on	P_stby
SC (Spacecraft)	135.26	0.00
GYRO_Sireus (GYRO Selex Galileo Sireus)	5.50	0.00
IMU_Astrix_1090A_1 (IMU Airbus Astrix 1090A #1)	21.00	0.00
IMU_Astrix_1090A_2 (IMU Airbus Astrix 1090A #2)	21.00	0.00
NavCam_1 (NavCam #1)	5.25	0.00
NavCam_2 (NavCam #2)	5.25	0.00
RW_HR04_1 (RW Honeywell HR04 #1)	9.60	0.00
RW_HR04_2 (RW Honeywell HR04 #2)	9.60	0.00
RW_HR04_3 (RW Honeywell HR04 #3)	9.60	0.00
RW_HR04_4 (RW Honeywell HR04 #4)	9.60	0.00
STR_HydraEU_Juice_1 (STR Sodern Hydra JUICE Electronics Unit #1)	11.55	0.00
STR_HydraEU_Juice_2 (STR Sodern Hydra JUICE Electronics Unit #2)	11.55	0.00
STR_HydraOH_Juice_1 (STR Sodern Hydra JUICE Optical Head #1)	7.88	0.00
STR_HydraOH_Juice_2 (STR Sodern Hydra JUICE Optical Head #2)	7.88	0.00
Grand Total	135.26	0.00

Table 11-8: AOCS mass and power budgets – extracted from OCDT

11.1.5 Options

There have been several trades made in this chapter whose outcomes have scope for re-evaluation if Ice Giants moves to phase A. These are summarised in the list below:

1. Safe mode strategy
    a. Baseline: star tracker based Earth-acquisition
    b. Alternative: Sun sensor + slew search to re-establish communications (as per NASA outer solar system s/c heritage)
2. Safe mode star tracker blinding mitigation strategy

   a. Baseline: low cost coarse rate sensor to initiate slew until star tracker un-blinded
   b. Alternative 1: a third IMU (just for safe mode) of same type as used in nominal mode to initiate slew until star tracker un-blinded
   c. Alternative 2: embarking a total of 4 optical heads evenly spaced in ring orthogonal to comms antenna, to guarantee that at least 1 head will see cold sky even in event of a single head failure
3. Attitude between science and communications period
   a. Baseline: Neptune or moon limb pointing of navigation camera
   b. Alternative: communications attitude for improved safety
4. Cruise mode attitude profile
   a. Baseline: spin-stabilised with all equipment off, with exception of 3-axis pointing to Earth for ~annual system checkout events.
   b. Alternative 1: spin-stabilised with all equipment off but slewing spin-axis regularly to maintain alignment of high-gain antenna with Earth
   c. Alternative 2: 3-axis stabilised pointing of high-gain antenna to Earth at all times, using reaction wheels
   d. Alternative 3: 3-axis stabilised pointing of high-gain antenna to Earth at all times, using thrusters only
5. Attitude stabilisation during main engine operations
   a. Baseline: 3-axis stabilised
   b. Alternative: spin-stabilised
6. Star tracker hardware use in safe mode
   a. Baseline: use redundant optical head and electronics unit in safe mode
   b. Alternative: embark a dedicated extra optical head and electronics unit for use in safe mode only
7. Fine pointing actuator
   a. Baseline: smallsat reaction wheels (1 Nms)
   b. Alternative 1: RCS default thrusters (10 N)
   c. Alternative 2: RCS monoprop thrusters (1 N)
   d. Alternative 3: cold gas thrusters
8. Number of RCS thrusters
   a. Baseline: 2 x 8 thruster strings
   b. Alternative 1: 2 x 4 thruster strings
   c. Alternative 2: 2 x 12 thruster strings

There are also several options for suppliers of the AOCS units. Some examples have been provided in sections 11.1.3.3 and 11.1.3.4 but the listed options are not exhaustive.

11.1.6 Technology Needs

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
X	RW_HR04	Reaction wheels (1 Nms)	Honeywell (US) or	7		May require parts upgrade to improve lifetime (5 years) and radiation hardness (20 krad is listed)
	RW	Reaction wheels (1 Nms)	Rockwell Collins Deutschland (Germany)	7/8?		Limited information on mass/power, lifetime, radiation, etc. for their 1 Nms wheels. Not clear if product line active and whether wheels can meet lifetime or radiation requirements of Ice Giants. Possible tech development to meet Ice Giant needs.

Table 11-9: possible AOCS technology development needs for Ice Giants

12 NEPTUNE GNC

12.1 Atmospheric Probe

12.1.1 Requirements and Design Drivers

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
AOCS-010	The GNC system shall trigger safely the various events of the Entry and Descent (E&D) sequence	
AOCS-020	The GNC system shall be able to record accelerations and angular rates for trajectory reconstruction. Note: the reconstruction is performed in post-processing not on-board	
AOCS-030	The GNC system shall be able to perform calibration of the acceleros and gyros before the entry phase.	

12.1.2 Assumptions and Trade-Offs

Assumptions	
1	The GNC will be passive during the ballistic flight after separation and before power up for the entry phase. Note: this means that there is no active control of the trajectory and entry conditions.
2	There are no specific performances of the IMU outputs required by the science community to be able to reconstruct the trajectory with a certain level of accuracy with respect to the centre of mass of the planet.
3	The separation strategy is similar to the Huygens' one, including the separation mechanism (RD[26] and RD[27]).

A trade-off considering having on-board navigation capability or no navigation has been performed.

- On board navigation: will reduce the amount of data to be transmitted for trajectory reconstruction and will provide more precise on-board knowledge of trajectory and attitude state. The drawback is the increased complexity in the on-board software (the GNC application software will include a full navigation function).
- No autonomous navigation: very simple GNC SW but the amount of data to be transmitted is higher (in order to be able to reconstruct the trajectory completely the raw accelerations and angular rates along with ancillary data needs to be transmitted).

For this mission it has been decided at system and customer level to keep a simple GNC SW and not perform any on-board navigation (including the calibration).

12.1.3 Baseline Design

12.1.3.1 Entry corridor analysis

In order to define separation requirements that ensure fulfilment of entry corridor conditions some analysis is performed. In this analysis it is important to note that there is a long flight time after separation (20 days at least was considered since the beginning of the study) and that there are no actuators on the probe (after separation the probe is in ballistic flight)

The review of the Huygens mission reveals that the total B-plane error is about 75 km (1σ) and the main contributions (which defines the Flight Path Angle corridor of ±1 deg 1-sigma) are:

- Orbit Determination error (~3 cm/s)
- Separation mechanism (~3 cm/s)
- Total pointing accuracy (~2 deg).

The assumed conditions for the entry corridor analysis are in the Neptune case:

- V infinity = 11.3 km/s
- Radius EIP = 25.690 km
- V separation mechanism = 0.4 m/s.

For an inertial FPA of -20 deg, an entry angle corridor of 1 deg (1-sigma) transforms into a B-plane error of 350 km (1-sigma). Assuming half of it (in the RSS sense) goes to orbit determination error and that the rest is shared equally between the pointing accuracy (APE) and the separation mechanism dispersion, there is plenty of margin to achieve the entry corridor even for further separation time (see pictures below).

Note that the arrival velocity has been assumed very high. In case this velocity is reduced the situation in terms of entry angle corridor improves (larger errors are allowed to achieve the same entry angle error).

For a FPA of -35 deg, the situation again is better (larger errors at separation can be tolerated to achieve the entry corridor).

Reducing the entry angle corridor to 0.1 deg, essentially decreases the tolerated errors by 10 times. Note that it might still be possible to achieve such demanding entry corridor with proper apportionment of the error sources. However, at this stage is considered not necessary to reach such accurate entry angle dispersion.

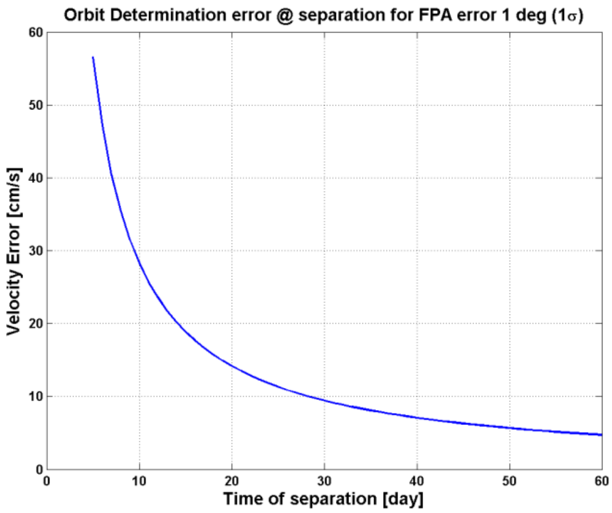

Figure 12-1: Orbit Determination velocity error (FPA -20 deg)

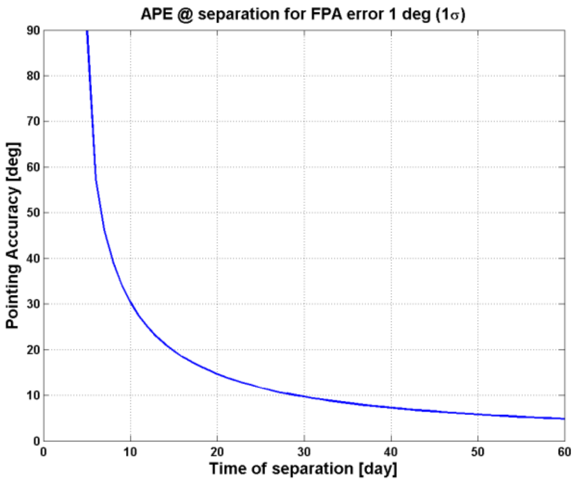

Figure 12-2: Pointing accuracy at separation (FPA -20 deg)

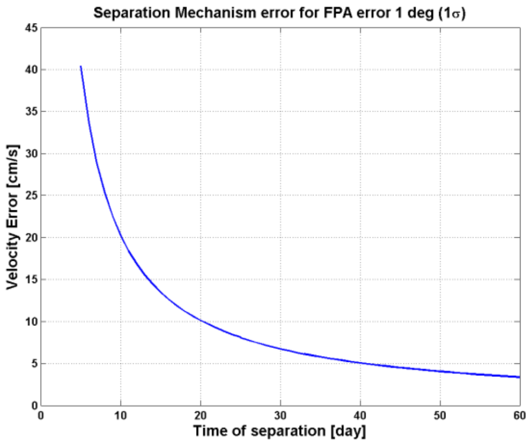

Figure 12-3: Separation mechanism transversal velocity error (FPA -20 deg)

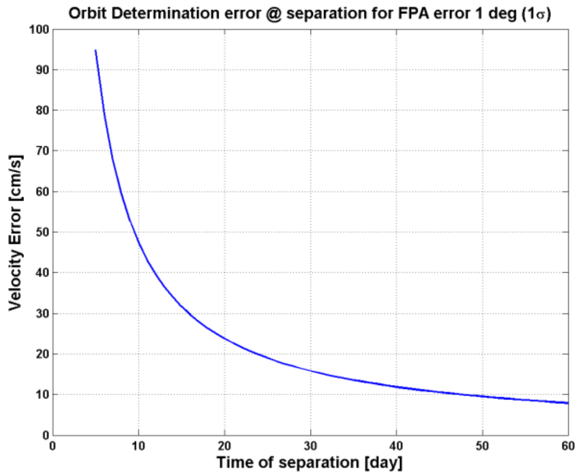

Figure 12-4: Orbit Determination velocity error (FPA -35 deg)

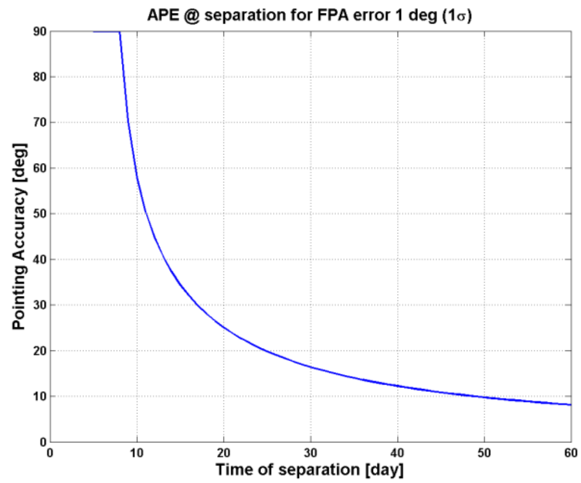

Figure 12-5: Pointing accuracy at separation (FPA -35 deg)

12.1.3.2 Calibration

In order to obtain data when the probe is not disturbed by the atmosphere, it is desirable to have the IMU powered ON about 10-20 min before EIP to enable calibration. This bias calibration is part of the post-processing.

Note that the IMU is 3-axis accelerometer and 3-axis gyro package (no compensation of accelero measurements due to rotation rate or angular acceleration).

12.1.4 List of Equipment

TheIMU (LN-200S) is the same as was used in PEP (RD[7]) is maintained due to heritage in multiple missions. However, it must be noted that it is ITAR-restricted and the radiation limit is 10 krad. The mass is 750 g.

As back-up of acceleros for parachute deployment a G-switch is proposed. The mass is 50 g. There is not much information available at the moment but it is heritage from Huygens.

	mass (kg)	mass margin (%)	mass incl. margin (kg)
Probe (Probe)	**1.60**	**5.00**	**1.68**
DM (Descent Module)	**1.60**	**5.00**	**1.68**
LN200S_1 (LN200S #1)	0.75	5.00	0.79
LN200S_2 (LN200S #2)	0.75	5.00	0.79
PAS_switch_1 (PAS Switch #1)	0.05	5.00	0.05
PAS_switch_2 (PAS Switch #2)	0.05	5.00	0.05
Grand Total	**1.60**	**5.00**	**1.68**

Table 12-1: The mass of the selected GNC equipment

Power (W)	P_on	P_stby
Probe (Probe)	**32.00**	**0.00**
DM (Descent Module)	**32.00**	**0.00**
LN200S_1 (LN200S #1)	16.00	0.00
LN200S_2 (LN200S #2)	16.00	0.00
PAS_switch_1 (PAS Switch #1)	0.00	0.00
PAS_switch_2 (PAS Switch #2)	0.00	0.00
Grand Total	**32.00**	**0.00**

Table 12-2: The power of the selected GNC equipment

12.1.5 Options

There are some European ‘IMU’ alternatives under development:

- MEMS gyro + MEMS acceleros (low performance, low mass) from TAS-UK
- Mini-FOG + quartz pendulum acceleros (high perform, higher mass) from Innalabs in Ireland.

12.1.6 Technology Needs

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
	IMU	Rad-hard, low-power, low-mass acceleros and gyros	TAS-UK, Innalabs (Ireland)	5	GSTP	Current technology for EDL IMU (Astrix 1090A) is 5 times more massive and power hungry but more accurate. Radiation hardening, bias stability during extended temperature and dynamics conditions

* Tick if technology is baselined

13 NEPTUNE POWER

13.1 Atmospheric Probe

13.1.1 Requirements and Design Drivers

The requirements that effectively drive the EPS subsystem design are best described by considering the Probe mission timeline:

- The EPS shall survive 13 years transfer attached to NASA orbiter. During this period, the probe will be in a normally dormant state, but some periodic activity for e.g. systems check-outs is foreseen.
- The EPS shall support, immediately after separation from the orbiter, 10 minutes of intermediate (PIM) mode (for systems check out and calibration etc.).
- The EPS shall then support 20 days of independent cruise in a dormant cruise (PCM) mode with all systems off except a timer (MTU).
- EPS shall then support, immediately before atmospheric entry, 36 minutes of intermediate (PIM) mode (for systems check out and calibration etc.).
- EPS shall then support 90 minutes of active descent (PDM) mode (in which the science mission is performed).

The power/energy requirements in each of the aforementioned systems modes are detailed below in Table 13-1 and Table 13-2. The energy requirements in Table 13-2 are derived according to the assumption that the probe will be exclusively battery powered following its release from a carrier spacecraft.

The final energy requirements in Table 13-2 include a system margin of 20%, an energy reserve of 20% (i.e. a battery depth-of discharge of 80%), and a 10% allowance for battery string redundancy.

13.1.1.1 Power/energy requirement budget

Row Labels	P_mean PDM	PCM	PIM
Probe (Probe)	362.1	0.0	330.5
COM	271.7	0.0	271.7
UHF_SSPA (UHF Solid State Power Amplifier)	133.3	0.0	133.3
UHF_SSPA_2 (UHF Solid State Power Amplifier #2)	133.3	0.0	133.3
UHF_TX (UHF Transmitter)	2.5	0.0	2.5
UHF_TX_2 (UHF Transmitter #2)	2.5	0.0	2.5
DH	3.5	0.005	5.0
CDM (Computer and Data Management Probe)	3.5	0.005	5.0
INS	38.4	0.0	5.4
ASI (Atmospheric Structure Instrument)	6.0	0.0	0.8
Cam_Rad (Camera-Radiometer)	9.6	0.0	1.3
Mass_Spec (Mass Spectrometer)	9.6	0.0	1.3
Phot (Photometer)	1.2	0.0	0.2
USO_Doppler (USO-Doppler)	12.0	0.0	1.7
MEC	0.0	0.0	0.0
BSSM_DM (Back Shell Separation Mechanism [DM side])	0.0	0.0	0.0
FSSM_DM (Front shield sep Mec [DM side])	0.0	0.0	0.0
Pyro_1 (Pyro #1)	0.0	0.0	0.0
Pyro_2 (Pyro #2)	0.0	0.0	0.0
Pyro_3 (Pyro #3)	0.0	0.0	0.0
PWR	16.5	0.0	16.5
PCDU_Pr (Power Conditioning & Distribution Unit_Probe)	16.5	0.0	16.5
GNC	32.0	0.0	32.0
LN200S_1 (LN200S #1)	16.0	0.0	16.0
LN200S_2 (LN200S #2)	16.0	0.0	16.0
Grand Total	**362.1**	**0.005**	**330.5**

Table 13-1: Probe power requirement budget at equipment-level. Values are time-averaged power in watts, and include an equipment maturity margin

Element properties		Level 3			
Element Definition short name:		Probe			
Element Definition long name:		Probe			
		PDM	PCM	PIM	
Total pwr incl. maturity margin (W)		**362.1**	**0.005**	**330.5**	
Harness Losses (W)	2%	7.2	0.000	6.6	
System Margin (W)	20%	72.4	0.001	66.1	
Total average power inc. Margin		**441.7**	**0.006**	**403.3**	
Mode duration (minutes)		90	28800	46	
Energy requirement (Wh)		662.6	2.9	309.2	**SUM TOTAL**
Energy req. incl 5% power conditioning loss (Wh)		**697.5**	**3.1**	**325.4**	**1026**
		incl.20% energy reserve			1231 Wh
		incl. 10% battery string redundancy			1354 Wh

Table 13-2: Summarised power and energy requirement budget (probe)

13.1.2 Assumptions and Trade-Offs

Assumptions	
1	The probe can receive (a small amount of) power from the carrier orbiter during transfer. This introduces an associated requirement upon the EPS: the EPS shall provide a power interface to the carrier orbiter.
2	Use of USA-provided radioisotope heater units (RHUs) will maintain EPS components at “normal” temperatures (approx. 10°C assumed) during transfer and cruise, without requiring use of electrical heating.

Table 13-3: Probe EPS assumptions

The major trade-off decision is between a primary (e.g. $LiSO_2$) battery, and a secondary rechargeable battery (Li-Ion). The trade-off is summarised in Table 13-4.

Trade consideration	Li primary	Li-ion secondary
Specific energy (cells @ BOL)	235 Wh/kg for LiSO2 with MER Heritage (@ 20°C). BUT much lower at lower temperatures (~26% less at 0°C).	TRL9: 140-170 Wh/kg @ 20°C but advancing quickly. Less affected by low temperature: 12% less capacity at 0°C w.r.t. 20°C.
Degradation during transfer?	Self discharge ~3% per year means < 160 Wh/kg after 13 years.	Calendar ageing. Very low for certain cell types (NCA). E.g. ~ 3% total capacity loss after 15 years @ 20°C.
Management during transfer?	Keep electrically isolated (but temp-controlled).	Keep at low SoC to minimise degradation. Temp must be controlled (capacity loss is much higher above 20°C).
Management during science mission?	Depassivation is needed before probe separation: this capability is needed in the PCDU (successful Huygens heritage).	Charge battery to 100% immediately before probe separation. Power from host spacecraft will be required. BCR needed within PCDU.

Table 13-4: Probe battery trade-off

In terms of battery mass, the trade is close. In the previous PEP CDF study (RD[7]), the selection was Li primary, specifically $LiSO_2$. This avoids the battery charge requirement with power from the host spacecraft. However, considering that a power and data interface to the probe would be needed in any case to allow for periodic check-out in transfer, this advantage is negligible.

Low temperatures (< 20°C) during the science mission will tend to push the trade towards Li-ion secondary due to the comparatively smaller reduction in secondary cell performance.

For the Ice Giants study (probe), the decision is to baseline a rechargeable Li-ion battery, because the EoL specific energy is slightly better, and temperature requirements are more flexible, especially the greater tolerance of lower temperatures during discharge.

13.1.3 Baseline Design

13.1.3.1 Battery

The battery mass and volume is sized assuming next-generation (but already high TRL) small-format space Li-ion cells with "NCA" (nickel-cobalt-aluminium) positive electrode chemistry for very low calendar ageing. A BoL nameplate specific energy of 169 Wh/kg is assumed.

For configuration reasons only, the battery is implemented as four separate battery units. The mass and size calculation, based on an EoL energy requirement of 1354 Wh (see Table 13-2) is shown in Table 13-5.

	BOL at cell level @ 20 deg C	169	Wh/kg
	Temperature reduction factor	0.94	at 10 deg C
	BOL at cell level @ mission temp	159	Wh/kg
	Packaging factor cells-to-battery	1.26	
	BOL at battery level	126	Wh/kg
	Calendar plus cycling degredation	0.995	per year
	Mission duration	15	years
	EOL at cell level	147	Wh/kg
	EOL at battery level	117	Wh/kg
	Battery density	0.92	g/cc
TOTAL all batteries	Battery mass	11.6	kg
	Battery volume	12.6	litres
	Number of batteries	4	
	Battery mass	2.9	kg
	Battery volume	3.2	litres
	Battery Height	165	mm
	Battery width	196	mm
	Battery length	97	mm

Table 13-5: Probe battery mass and size calculation (based on assumption of small-format Li-ion NCA cells with nameplate specific energy of 169 Wh/kg)

13.1.3.2 PCDU

The PCDU should be quite simple, having to interface only with one energy source (battery). The maximum power delivery capability is modest (approx. 360 watts). Therefore, a reasonably small and light unit is foreseen. The mass and size estimates are based approximately on Medium Modular Power System by Terma A/S, with a selection of functionality appropriate to this case, as detailed in Table 13-6.

	Mass, kg per module	# of modules	Total mass, kg	
Equipment power distribution module	0.570	2	1.1	
Pyro firing module	0.476	2	1.0	
BCDR module	0.550	3	1.7	
"Power interface module" (mass of BDR module assumed)	0.575	2	1.2	
MIL1553 Interface module	0.458	2	0.9	
		Mass of all modules	**5.8**	kg
		Total mass of PCDU incl backplane and structure	**8.8**	kg

Width	0.235	metres
Height	0.156	metres
Length	0.279	metres
volume	10.2	litres
density	0.86	g/cc

Table 13-6: Probe PCDU mass and size estimates (based very approximately on *Medium Modular Power System* by Terma A/S)

13.1.4 List of Equipment

	mass (kg)	mass margin (%)	mass incl. margin (kg)
Bat_Pr (Battery_Probe)	2.90	5.00	3.05
Bat_Pr_2 (Battery_Probe #2)	2.90	5.00	3.05
Bat_Pr_3 (Battery_Probe #3)	2.90	5.00	3.05
Bat_Pr_4 (Battery_Probe #4)	2.90	5.00	3.05
PCDU_Pr (Power Conditioning & Distribution Unit_Probe)	8.71	10.00	9.58
Grand Total	**20.31**	**7.14**	**21.76**

Other parameters	TIDS	len	height	wid	P_on	P_stby	TRL
Bat_Pr (Battery_Probe)	4000	97	165	196			6
Bat_Pr_2 (Battery_Probe #2)	4000	97	165	196			6
Bat_Pr_3 (Battery_Probe #3)	4000	97	165	196			6
Bat_Pr_4 (Battery_Probe #4)	4000	97	165	196			6
PCDU_Pr (Power Conditioning & Distribution Unit_Probe)	50	279	156	235	16.5	16.5	5

Table 13-7: EPS Equipment list (Neptune Probe)

13.1.5 Options

A credible option would be to use a primary battery rather than a secondary rechargeable one. This is discussed in Section 13.1.2 above.

13.1.6 Technology Needs

No new technologies are required.

13.2 Orbiter

13.2.1 Requirements and Design Drivers

Requirements on the EPS, at this level of design, are dominated firstly by:

- Provision of power at 30 AU from the sun (or 19 for Uranus). This rules out solar power
- Total (autonomous phase) mission energy requirements of several MWh. This rules out any energy storage technology as the primary source.

Having therefore established that nuclear power is mission enabling, the EPS is not classically designed and sized according to imposed requirements, but rather the spacecraft system, mission and EPS subsystem are iterated in parallel, informed mainly by the high level assumptions regarding the details of the nuclear power source provision (see below).

13.2.1.1 Orbiter power requirement budget

Row Labels	TM	LEOP	CM	MM	ObM	ComM	NSM	IPCA	IMCA1	IMCA2	ISCom	IN
⊟ AOGNC	0.0	41.0	6.9	69.2	69.8	48.2	21.5					
IMU_Astrix_1090A_1 (IMU Airbus Astrix 1090A #1)	0.0	10.5	1.1	10.5	10.5	0.0	1.1					
IMU_Astrix_1090A_2 (IMU Airbus Astrix 1090A #2)	0.0	10.5	1.1	10.5	10.5	0.0	1.1					
NavCam_1 (NavCam #1)	0.0	0.3	0.0	0.0	0.3	0.0	0.0					
NavCam_2 (NavCam #2)	0.0	0.3	0.0	0.0	0.3	0.0	0.0					
RW_HR04_1 (RW Honeywell HR04 #1)	0.0	0.0	0.7	7.2	7.2	7.2	0.0					
RW_HR04_2 (RW Honeywell HR04 #2)	0.0	0.0	0.7	7.2	7.2	7.2	0.0					
RW_HR04_3 (RW Honeywell HR04 #3)	0.0	0.0	0.7	7.2	7.2	7.2	0.0					
RW_HR04_4 (RW Honeywell HR04 #4)	0.0	0.0	0.7	7.2	7.2	7.2	0.0					
STR_HydraEU_Juice_1 (STR Sodern Hydra JUICE Electronics Unit #1)	0.0	5.8	0.6	5.8	5.8	5.8	5.8					
STR_HydraEU_Juice_2 (STR Sodern Hydra JUICE Electronics Unit #2)	0.0	5.8	0.6	5.8	5.8	5.8	5.8					
STR_HydraOH_Juice_1 (STR Sodern Hydra JUICE Optical Head #1)	0.0	3.9	0.4	3.9	3.9	3.9	3.9					
STR_HydraOH_Juice_2 (STR Sodern Hydra JUICE Optical Head #2)	0.0	3.9	0.4	3.9	3.9	3.9	3.9					
⊟ COM	15.0	153.0	28.8	15.0	15.0	195.6	28.8					
KaEPC (Ka-Band Electronic Power Conditioning)	0.0	0.0	0.0	0.0	0.0	4.4	0.0					
KaEPC_RED (Ka-Band Electronic Power Conditioning - Redundant)	0.0	0.0	0.0	0.0	0.0	4.4	0.0					
KaTWT (Ka-Band Traveling Wave Tube)	0.0	0.0	0.0	0.0	0.0	84.5	0.0					
KaTWT_RED (Ka-Band Traveling Wave Tube - Redundant)	0.0	0.0	0.0	0.0	0.0	84.5	0.0					
XEPC (X-Band Electronic Power Conditioning)	0.0	2.9	0.3	0.0	0.0	0.1	0.3					
XEPC_RED (X-Band Electronic Power Conditioning - Redundant)	0.0	2.9	0.3	0.0	0.0	0.1	0.3					
XKa_XPND_RED (X/X/Ka-Band Transponder - Redundant)	7.5	17.5	8.5	7.5	7.5	7.7	8.5					
XKaXPND (X/X/Ka-Band Transponder)	7.5	17.5	8.5	7.5	7.5	7.7	8.5					
XTWT (X-Band Traveling Wave Tube)	0.0	56.0	5.6	0.0	0.0	1.1	5.6					
XTWT_RED (X-Band Traveling Wave Tube - Redundant)	0.0	56.0	5.6	0.0	0.0	1.1	5.6					
⊟ CPROP	3.5	3.5	19.3	256.7	19.3	19.3	19.3					
Biprop_LP_Trans_1 (Biprop_LP_Transducer)	0.8	0.8	0.8	0.8	0.8	0.8	0.8					
Biprop_LP_Trans_2 (Biprop_LP_Transducer)	0.8	0.8	0.8	0.8	0.8	0.8	0.8					
Biprop_LV_1 (Biprop_Latch_Valve)	0.0	0.0	0.6	1.5	0.6	0.6	0.6					
Biprop_LV_2 (Biprop_Latch_Valve)	0.0	0.0	0.6	1.5	0.6	0.6	0.6					
Biprop_LV_3 (Biprop_Latch_Valve)	0.0	0.0	0.6	1.5	0.6	0.6	0.6					
Biprop_LV_4 (Biprop_Latch_Valve)	0.0	0.0	0.6	1.5	0.6	0.6	0.6					
Biprop_Thruster_Main_1 (Biprop_Thruster_Main #1)	0.0	0.0	0.0	90.0	0.0	0.0	0.0					
Biprop_Thruster_Main_2 (Biprop_Thruster_Main #2)	0.0	0.0	0.0	90.0	0.0	0.0	0.0					
Biprop_HP_Trans (Biprop_HP_Transducer)	0.3	0.3	0.3	0.3	0.3	0.3	0.3					
Biprop_LP_Trans_3 (Biprop_LP_Transducer)	0.8	0.8	0.8	0.8	0.8	0.8	0.8					
Biprop_LP_Trans_4 (Biprop_LP_Transducer)	0.8	0.8	0.8	0.8	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_01 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_02 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_03 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_04 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_05 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_06 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_07 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_08 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_09 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_10 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_11 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_12 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_13 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_14 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_15 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					
Biprop_Thruster_RCS_1_16 (Biprop_Thruster_RCS #1)	0.0	0.0	0.8	4.2	0.8	0.8	0.8					

⊟ DH	63.0	63.0	37.8	63.0	63.0	63.0	25.2					
RIUC (Remote Inteface Unit Centralised)	16.0	16.0	9.6	16.0	16.0	16.0	6.4					
RIUD (Remote Interface Unit Decentralised)	12.0	12.0	7.2	12.0	12.0	12.0	4.8					
CDMU_1 (Computer and Data Management Unit #1)	17.5	17.5	10.5	17.5	17.5	17.5	7.0					
CDMU_2 (Computer and Data Management Unit #2)	17.5	17.5	10.5	17.5	17.5	17.5	7.0					
⊟ INS								145.1	165.7	218.2	120.8	18.0
Cam (Camera)								34.8	23.2	23.2	0.0	0.0
Im_spec (Imaging Spectrometer)								25.2	23.1	23.1	0.0	0.0
INMS (Ion and Neutral Mass Spectreometer)								0.0	0.0	17.5	0.0	0.0
INMS_2 (Ion and Neutral Mass Spectreometer #2)								0.0	0.0	17.5	0.0	0.0
INMS_3 (Ion and Neutral Mass Spectreometer #3)								0.0	0.0	17.5	0.0	0.0
Mag (Magnetometer)								12.0	12.0	12.0	12.0	12.0
Micro_rad (Microwave radiometer)								67.1	67.1	67.1	0.0	0.0
USO (Ultra Stable Oscillator)								6.0	6.0	6.0	6.0	6.0
Ins_KaEPC (Instrument Ka-Band Electronic Power Conditioning)								0.0	1.0	1.0	3.0	0.0
InsKaTWT (Instrument Ka Band Traveling Wave Tube)								0.0	20.0	20.0	60.0	0.0
Ka_Transp (Ka-band Trransponder)								0.0	13.3	13.3	39.8	0.0
⊟ PWR	24.0	24.0	24.0	24.0	24.0	24.0	24.0					
PCDU_Orb (Power Conditioning & Distribution Unit_Orbiter)	24.0	24.0	24.0	24.0	24.0	24.0	24.0					
⊞ TC	38.0	0.0	45.0	0.0	0.0	0.0	20.0					
⊟ RAD	2.7	0.3	2.7	0.0	2.7	2.7	2.7					
rad_mon_ngrm (Radiation Monitor NGRM)	2.7	0.3	2.7	0.0	2.7	2.7	2.7					

Table 13-8: Orbiter power requirement budget at equipment-level, according to system mode (platform and instruments). Values are time-averaged power in watts, and include an equipment maturity margin

"Mission mode" --->	Closest approach Neptune	Closest approach Triton (1)	Closest approach Triton (2)	Comms (duration per 24hr period)	Nominal Science	
Instruments sys. mode	IPCA	IMCA1	IMCA2	ISCom	IN	
Platform sys. mode	ObM	ObM	ObM	ComM	NSM	
Mission mode duration	81.3	22.0	2.0	4.5	Remainder	hrs
Instruments av. power	145	166	218	121	18	W
Instruments incl maturity margin	145	166	218	121	18	W
Platform av. power	194	194	194	353	142	W
Platform incl maturity margin	194	194	194	353	142	W
Total av. power incl maturity margin	**339**	**359**	**412**	**474**	**160**	W

Table 13-9: Summarised orbiter average power requirement budget according to "mission mode" (hybrid of platform and instrument system modes)

13.2.2 Assumptions and Trade-Offs

Assumptions	
1	Availability & provision of 3 eMMRTGs
2	EOM power output of one eMMRTG = 90 W

Table 13-10: Orbiter EPS assumptions

13.2.2.1 Assumption 1: availability & provision of 3 eMMRTGs

There are risks to this assumption as follows:

Pu-238 availability

U.S. stocks of Pu-238 radioisotope fuel have been reducing since cessation of manufacture in the 1980s. A new programme to restart production is now underway, but Pu-238 remains a scarce resource that is, in effect, competed for by different mission proposals.

The NASA Ice Giants report, (RD[1] section D.5.1) states: "as of 2016 DOE can fuel 4 generators including the one for the Mars 2020" [implying availability of 3 generators for an Ice Giants mission]. In order to provide fuel for an additional 2 RTGs for a total 5 RTGs [for Ice Giants], it would require approx. 6 additional years for fuel processing"

Furthermore, RD[28] states "DOE officials said they now expect to reach full [1.5 kg/yr] production no earlier than 2025 with a late completion date remaining in 2026".

It can be concluded that a mission concept including 3 or more RTGs for a NASA element plus 3 RTGs for a European element is dependent on both good performance of the DOE fuel programme, and prioritisation of the mission within the NASA strategy.

Ground facilities and integration

The NASA Ice Giants report [JPL D-100520] states: "no more than 4 RPS into a spacecraft are recommended". This is based on maximum storage capacity (both at INL and KSC) and availability of doors in launcher fairing. In addition, side-by-side configuration for the two complementary orbiters would not allow a late integration of RTGs. This point is one reason to favour a vertical stack rather than a side-by-side concept for a two-orbiter configuration.

13.2.2.2 Assumption 2: EOM power output of one eMMRTG = 90 W

The enhanced MMRTG (eMMRTG) is a new version of the device powering the MSL Curiosity rover – it uses a new type of thermoelectric couple, partly motivated by a requirement to decrease the power degradation rate. However, despite the foreseen improvement, the power output reduction from the eMMRTG will be very significant over a mission of ~15 years. Furthermore, because the eMMRTG is a new development, the long term performance characteristics have significant uncertainty.

Recent references give EODL end-of-design-life (nominally 3 years storage +14 year mission) power estimates in the range of 80 to 100W.

For this Ice Giants study, 90W EOM end-of-mission (15 years after launch) is assumed.

Most recently, in an abstract submitted to the 2019 IEEE Aerospace Conference, 77 W at EODL is mentioned as the requirement.

In conclusion, the 90 W at EOM assumption is subject to later refinement and/or confirmation. An uncertainty level of ~15% seems appropriate at the time of writing.

13.2.2.3 Major EPS trade-off

The most fundamental EPS design option is:

- To embark little or no secondary energy storage (rechargeable batteries), and therefore constrain the spacecraft system and mission power requirements to be always below the RTG power output.

This is the classical approach of USA deep space probes e.g. *Voyager*, *Cassini*, *New Horizons*. In some cases a large high-voltage capacitor bank provides for very-short-term power spikes (e.g. turn-on in-rush), but there is no battery.

Or,

- To run the mission functionality in a periodic way, from secondary batteries, at power levels exceeding the RTGs output. Interspersed with semi-dormant recharge periods.

  This is the approach of MSL Curiosity rover.

The power requirements of the science instruments and the communication subsystem, together with the constraints on number of RTGs available, lead to selection of the second option (battery supported).

13.2.2.4 EPS model

The power subsystem is modelled using ESA TEC-EP simulation platform *PEPS*. This allows dynamic modelling of full power system with a load profile of unlimited complexity, which is well suited to modelling the situation of the Neptune orbit with the complex periodical concept of operations.

The schematic representation of the power system model is shown in Figure 13-1. The model was used to find the required battery energy, and also the maximum duration of the communication mode that could be supported in a periodic way, once in every 24 hours.

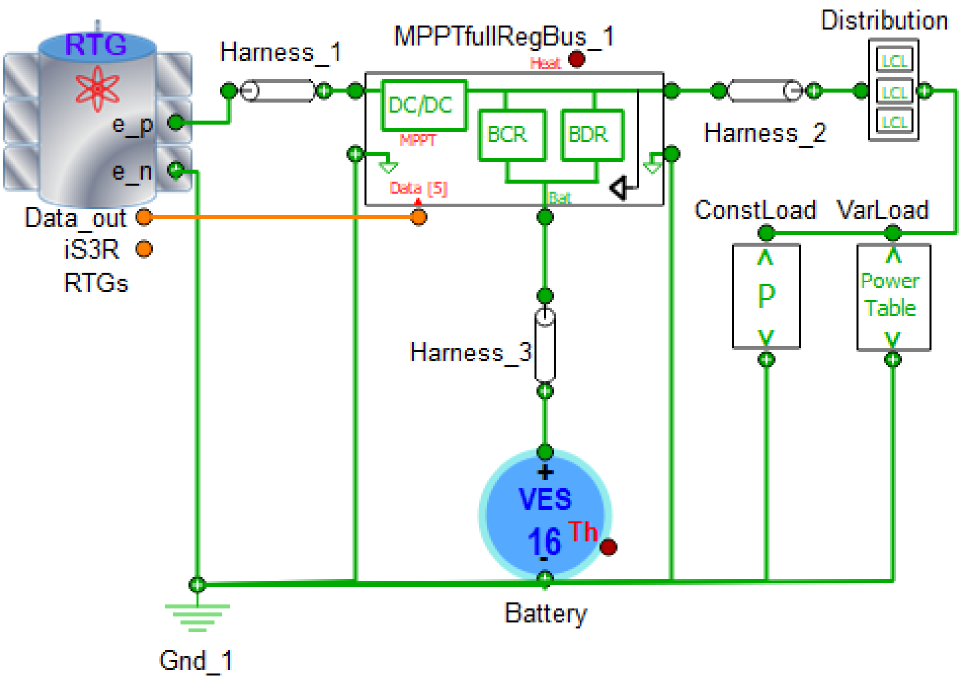

Figure 13-1: Schematic representation of the PEPS power system model for Ice Giants Orbiter

Figure 13-2 shows an example of the simulation results output. It begins with the Neptune close approach, in which the electrical load profile flip-flops between science and communications modes, relying on provision of stored energy and discharging the battery to 20% SOC. At 120 hours the load switches to flip-flop between Communications and Nominal Science profiles, which provides a slow battery recharge in a sawtooth pattern.

Figure 13-3 shows the EPS model results for one full 50-day orbit (Triton approach at ~650 hours): Battery is fully charged just before orbit completion, showing that the energy demand is maximised w.r.t. the energy available.

These results were obtained with a battery size of 19.8 kWh without redundancy (21.8 kWh including 10% string redundancy). The duration of the communications mode is 3.2 hours in every 24 hour period.

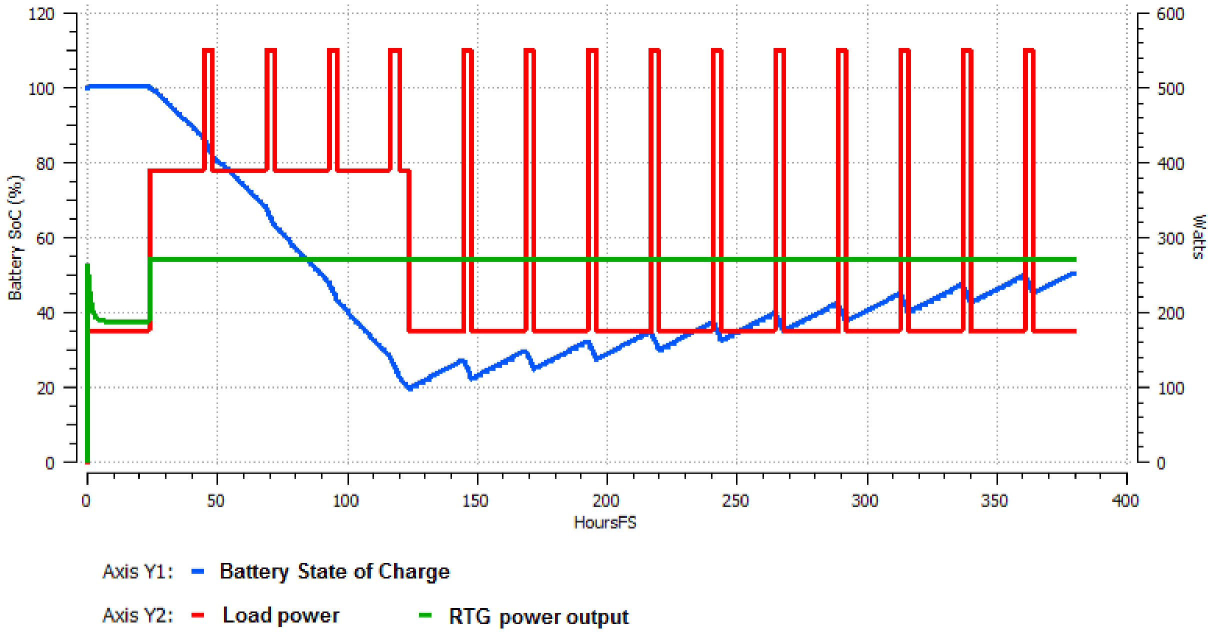

Figure 13-2: PEPS model results showing the Neptune approach (between 20 and 120 hours)

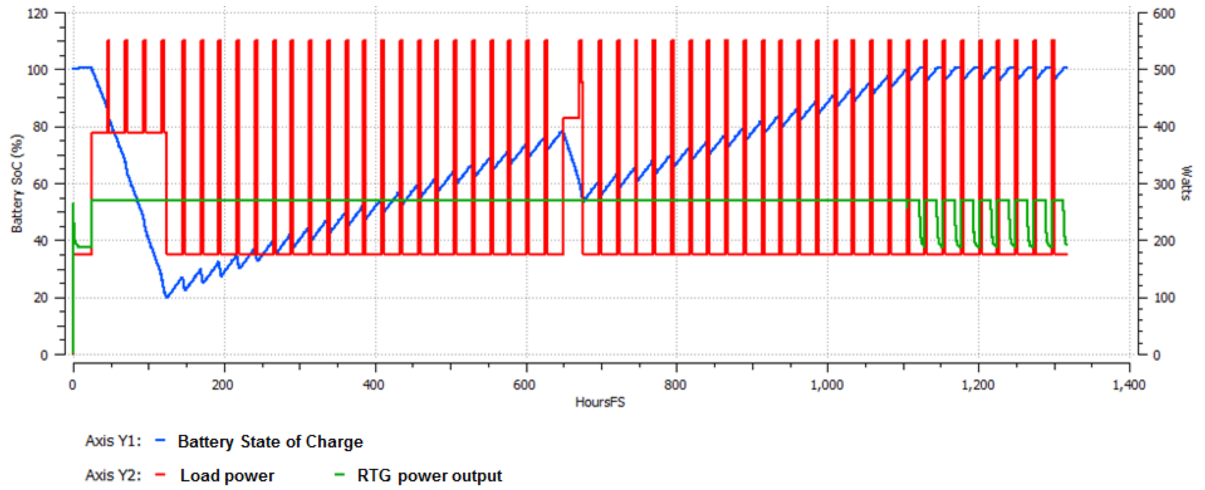

Figure 13-3: PEPS model results for 1 full 50-day orbit (Triton approach at ~650 hours)

13.2.3 Baseline Design

13.2.3.1 Battery

The battery is sized assuming new-generation (but already qualified) space large-format Li-ion cells with “NCA” (nickel-cobalt-aluminium) positive electrode chemistry for low calendar ageing. It is large, so is split into four modules.

	BOL cell energy	182	Wh
	Cell mass	1.079	kg
	BOL at cell level	169	Wh/kg
	packaging factor cells-to-battery	1.26	
	BOL at battery level	134	Wh/kg
	Calendar plus cycling deg	0.995	per year
	Mission duration	15	years
	EOL at cell level	156	Wh/kg
	EOL at battery level	124	Wh/kg
	Battery density	0.92	g/cc
TOTAL all batteries	Battery mass	176	kg
	Battery volume	190.884444	litres
	Number of batteries	4	
	Battery mass	43.9	kg
	Battery volume	48	litres
	Battery Height	265	mm
	Battery width	235	mm
	Battery length	766	mm

Table 13-11: Probe battery mass and size calculation (based on assumption of large-format Li-ion space-qualified cells)

13.2.3.2 PCDU

Mass and size estimation is based very approximately on the Medium Modular Power System from TERMA A/S, with functionality tailored to this case.

	Mass, kg per module	# of modules	Total mass, kg	
Equipment power distribution module	0.570	2	1.1	
Pyro firing module	0.476	1	0.5	
"RTG power control module" (mass of an APR MPPT module assumed)	0.500	4	2.0	
BCDR module	0.550	2	1.1	
"Power interface module" (mass of BDR module assumed)	0.575	2	1.2	
MIL1553 Interface module	0.458	2	0.9	
		Mass of all modules	**6.78**	kg
		Total mass of PCDU incl backplane and structure	**10.29**	kg

Width	0.235	metres
Height	0.156	metres
Length	0.329	metres
volume	12.1	litres
density	0.85	g/cc

Table 13-12: Orbiter PCDU mass and size estimates (based very approximately on *Medium Modular Power System* by Terma A/S)

13.2.3.3 Resistive power shunts

RTGs must be subject to a reasonably constant load (close to the maximum power point) in order to maintain internal temperatures within specification. Therefore, the power conditioning system must include resistive shunts for the dissipation of excess power whenever is it not used by the spacecraft electrical equipment. It is assumed that both internal and external resistive shunts are included, for spacecraft thermal management reasons.

13.2.3.4 RTGs

Three USA eMMRTGs are baselined. See Figure 13-4 and Table 13-13.

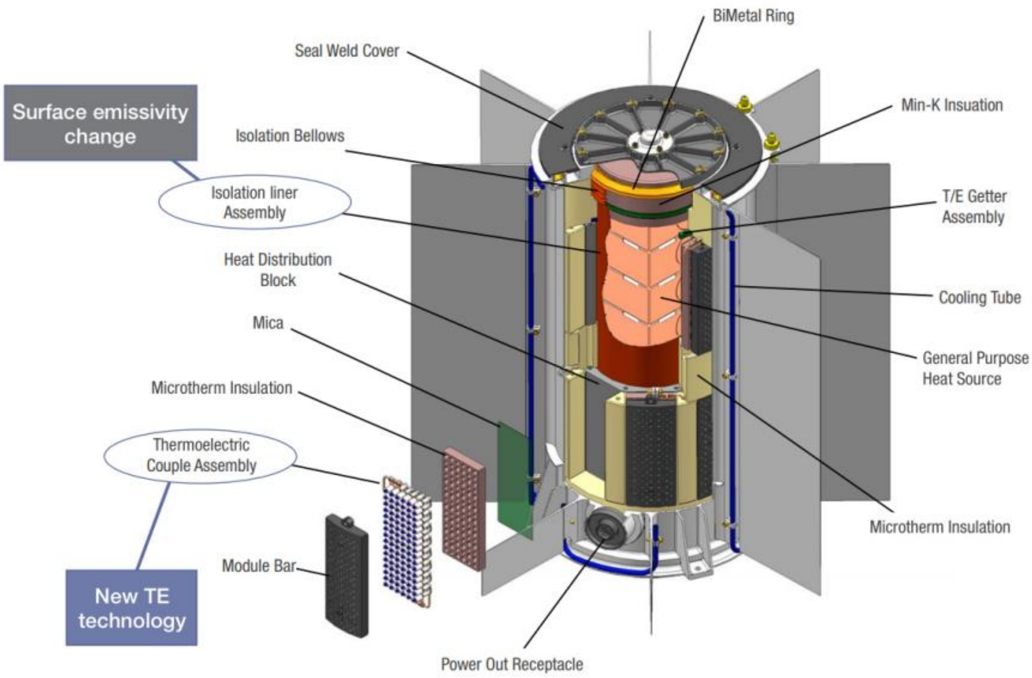

Figure 13-4: eMMRTG – courtesy of NASA

Property	Value
Diameter (fin tip to tip)	0.65 m
Length	0.69 m
Mass	45 kg
BOM power	145 W_{el}
Estimated EODL power *3-year storage + 14-year mission*	> 90 W_{el}
BOL specific power	~3.5 W_{el}/kg
Power degradation rate	2.5% /year
Allowable flight voltage envelope	22-34 V
Heat rejection in vacuum BOM	1854 W_{th}
Heat rejection in vacuum EODL	1649 W_{th}
Fin root temperature in deep space	420 K
Max allowable fin root temperature	473 K

Table 13-13: eMMRTG characteristics

13.2.4 List of Equipment

	mass (kg)	mass margin (%)	mass incl. margin (kg)
Bat_Orb (Battery_Orbiter)	43.9	5.0	46.1
Bat_Orb_2 (Battery_Orbiter #2)	43.9	5.0	46.1
Bat_Orb_3 (Battery_Orbiter #3)	43.9	5.0	46.1
Bat_Orb_4 (Battery_Orbiter #4)	43.9	5.0	46.1
EMMRTG (Enhanced_Multi_Mission_RTG)	45.0	10.0	49.5
EMMRTG_2 (Enhanced_Multi_Mission_RTG #2)	45.0	10.0	49.5
EMMRTG_3 (Enhanced_Multi_Mission_RTG #3)	45.0	10.0	49.5
Ext_Pwr_Shnt (External power shunt)	1.0	20.0	1.2
PCDU_Orb (Power Conditioning & Distribution Unit_Orbiter)	10.3	20.0	12.4
Res_Pwr_Shnt (Resisitive power shunt)	1.0	20.0	1.2
Res_Pwr_Shnt_2 (Resisitive power shunt #2)	1.0	20.0	1.2
Res_Pwr_Shnt_3 (Resisitive power shunt #3)	1.0	20.0	1.2
Grand Total	**324.9**	**7.7**	**350.0**

Other parameters	TIDS	len	diam	height	wid	P_on	P_stby	TRL
Bat_Orb (Battery_Orbiter)	4000	766	0	265	235			7
Bat_Orb_2 (Battery_Orbiter #2)	4000	766	0	265	235			7
Bat_Orb_3 (Battery_Orbiter #3)	4000	766	0	265	235			7
Bat_Orb_4 (Battery_Orbiter #4)	4000	766	0	265	235			7
EMMRTG (Enhanced_Multi_Mission_RTG)		690	650	0	0			4
EMMRTG_2 (Enhanced_Multi_Mission_RTG #2)		690	650	0	0			4
EMMRTG_3 (Enhanced_Multi_Mission_RTG #3)		690	650	0	0			4
Ext_Pwr_Shnt (External power shunt)	0	200	0	10	40	0	0	5
PCDU_Orb (Power Conditioning & Distribution Unit_Orbiter)	50	329	0	156	235	24	24	4
Res_Pwr_Shnt (Resisitive power shunt)	0	200	0	10	40	0	0	5
Res_Pwr_Shnt_2 (Resisitive power shunt #2)	0	200	0	10	40	0	0	5
Res_Pwr_Shnt_3 (Resisitive power shunt #3)	0	200	0	10	40	0	0	5

Table 13-14: EPS Equipment list (Neptune Orbiter)

13.2.5 Options

13.2.5.1 Number of RTGs

An analysis was performed to determine if the mission was feasible with only 2 eMMRTGs.

The energy budget could not be balanced, even with severe restriction of mission functionality (e.g. communication link time). Therefore this option is rejected.

13.2.6 Technology Needs

A PCDU to interface with eMMRTGs will need to be designed, manufactured and qualified, but this involves no unknown factors and is not a new technology development in the sense intended here.

As mentioned above, the eMMRTGs are still in development (with high current TRL), but this aspect is taken care of on the NASA side.

14 NEPTUNE TELECOMMUNICATIONS

14.1 Atmospheric Probe

14.1.1 Requirements and Design Drivers

The requirements for the telecommunication subsystem are shown below:

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
COM-010	The telecommunication subsystem shall be able to receive a telemetry (TM) data stream from the data handling system and to transmit this data to the Orbiter.	
COM-020	The Probe-to-Orbiter link shall adopt a residual carrier signal. *Rationale: a residual carrier signal has simpler acquisition and tracking with respect to suppressed carrier signals, and it allows to perform signal detection by Earth Radiotelescopes for Probe aliveness.*	
COM-030	The Probe shall be able to transmit all generated data from instruments (payload telemetry) and the system (housekeeping telemetry) to the Orbiter.	
COM-040	The transmitter function shall be hot redundant.	

The main design drivers for the communication subsystem can be identified in the followings:

- *The required bitrate*: COM-030 requires to size the bitrate to a minimum value so that all generated data can be transmitted to the Orbiter. This will drive the minimum radiofrequency (RF) power for transmission, and thus the sizing of the power subsystem.
- *Orbiter antenna and pointing capabilities*: for meeting the minimum bitrate (COM-030), the Orbiter antenna and pointing capabilities will drive the antenna sizing of the probe, and the minimum RF power for transmission (and thus also the sizing of the power subsystem).

14.1.2 Assumptions and Trade-Offs

Assumptions	
1	The Orbiter antenna has an antenna aperture > $1.5^2\pi$ [m] that is pointed in the Probe direction.
2	The worst case noise temperature seen by the orbiter receiver, caused by Neptune, is 1000 K.
3	The worst case atmospheric losses experienced by the Probe-to-Orbiter link is 10.6 dB.
4	The required bitrate for transmitting all generated data (COM-030) is 2 kbps
5	The change of transmitter (due to possible failure) is recognized by the FDIR.

Concerning Assumption 1, As the NASA Orbiter communication subsystem design (for the Probe-to-Orbiter link) is out of the scope of the CDF study, it has been assumed that the minimum antenna aperture (that drives the Orbiter antenna dimension) is (greater than or) equal to $1.5^2\pi$ m that corresponds (for instance) to the following antenna gains:

- ~14.5 dB at UHF, 435-450 MHz,
- ~29 dB at S-Band, 2290 MHz,
- ~40 dB at X-Band, 8450 MHz.

Concerning Assumption 2 and 3, they have been derived by the CDF study PEP (RD[7]), for which a preliminary estimation of noise temperature and atmospheric losses on Neptune was done. No re-assessment of such values has been done during the CDF IceGiants, and the consolidation of such values should be done in next study phases.

Assumption 4 comes from the computation done at system level of the data volume against transmission time. In particular, based on the input provided by Mission analysis, a minimum bitrate of 2 kbps was derived.

Finally, concerning Assumption 5, it has been assumed that the FDIR can recognize possible failures of the transmitter, and hence it is able to switch to the redundant one without the need of having a receiving link (Orbiter-to-Probe).

14.1.2.1 Frequency allocation trade-offs

During the CDF study, a trade-off on the following frequency bands was considered:

- UHF, 435-450 MHz,
- S-Band, 2200-2300 MHz,
- X-Band, 8025-8500 MHz.

While the S- and X-Band could limit antenna dimensions, the UHF has the following advantages:

- Solid state power amplifier (SSPA) with RF output power up to 80 W already qualified (BioMass heritage)
- Better omni-directional coverage of the antenna (or array of antennas) on the Probe side
- Huygens and ExoMars heritage concerning UHF transmitters
- Typically lower mass of units and components w.r.t. S- and X-Band ones
- Compatibility with UHF Radiotelescope arrays for Probe aliveness signal detection.

With reference to the first advantage, Table 14-1 shows a link budget comparison between UHF and X-Band, considering the best amplifier option for both cases. In particular, the link budget assumes:

- 80 W for UHF, 12 W for X-Band
- The worst case distance (Probe-to-Orbiter)
- An LGA on the probe side
- An (equivalent) antenna aperture of 1.5 m on the Orbiter side.

From the link budget, it can be seen that UHF, thanks to the SSPA heritage, allows to achieve the required 2 kbps with a low gain antenna (LGA) on the Probe (that maximises antenna coverage). Hence, UHF has been selected for the baseline design.

PARAMETER	UHF	X-Band	Notes
RANGE [km]	38000.0	38000.0	
FREQUENCY [MHz]	450	8450	
TX POWER [W]	80	11.99	**Amplifier heritage**
TX ANTENNA GAIN [dB]	-0.63	-0.63	LGA
TX LOSSES [dB]	1	1	
TX EIRP [dBW]	**17.40**	**9.16**	**Calculated**
PATH LOSSES [dB]	**177.10**	**202.57**	**Calculated**
ATMOSPHERE LOSS [dB]	10.65	12.00	Best estimation
RX G/T [dBK]	-19.00	6.89	Based on Assumption 1 and 2
DEMOD. LOSS [dB]	0.80	0.80	Estimation
MOD. LOSS [dB]	0.61	0.61	Residual carrier modulation
REQUIRED Eb/No [dB]	1.80	1.80	LDPC coding
MINIMUM MARGIN [dB]	3.00	3.00	Standard ESA
MAX BIT RATE [dBHz]	33.04	23.86	
MAX BIT RATE [kbps]	**2.01**	**0.24**	

Table 14-1: UHF versus X-Band trade-off for the probe, in terms of achievable bitrate

14.1.3 Baseline Design

The baseline design of the Probe UHF communication subsystem foresees an architecture as shown in Figure 14-1 and includes:

- Two UHF transmitters
- Two SSPAs
- One array of LGA UHF antennas (on the Probe backshell) patch-like providing a low gain coverage
- One UHF LGAs
- The RFDN that interconnects all the aforementioned devices.

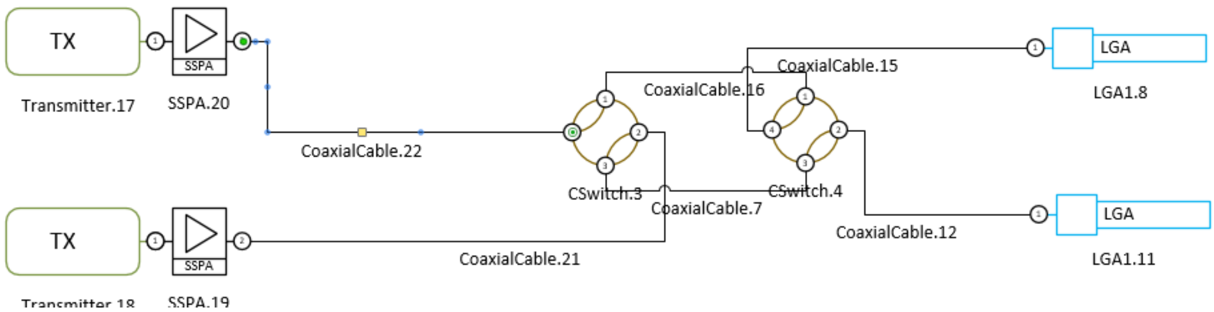

Figure 14-1: Baseline design of the Probe communication subsystem

Of the two transmitter, only one is adopted for nominal operation. The second transmitter is used for redundancy in hot mode (see requirement COM-020). The transmitter is able to modulate a TM signal, SP-L/PM, 4 ksps. The coding function (that shall be implemented in the DHS) is LDPC, hence the supported net bit rate is 2 kbps.

The two SSPA bring the transmitted signal at the required RF output power, i.e. 80 W that allows the orbiter to demodulate the received signal with a frame error rate of 1e-5 (one frame lost every 100,000). Additionally, the transmitted signal carrier can be detected by an array of 20 radio telescopes on Earth.

The array of UHF antennas on the Probe backshell is designed to provide an almost omnidirectional coverage between -80 and +80 degrees from the boresight. A preliminary concept design of such antenna configuration is provided in Figure 14-2.

After backshell ejection, the communication subsystem routes the transmitted signal to the second LGA. Also this antenna can provide an omnidirectional coverage between -80 and +80 degrees from the boresight.

The output TM signal from the active transmitter can be routed by means of the RFDN to the two LGAs. During the CDF study, two switches and coaxial cables were considered so that the two transmitters are fully cross-strapped with the two LGAs. However, other RFDN could be investigated during next phases by trading off reliability and cost versus mass and configuration.

Finally, it is pointed out that:

- The baseline design assumes that the FDIR can recognize possible failures of the active transmitter (Assumption 5), hence removing the need of having receivers. During next phases, the feasibility of such approach shall be assessed more in detail, since the need of receivers implies a major re-design of the Probe (impact on mass, volume, and thus accommodation of the instruments).
- The baseline design also assumes that a frame error rate of 1e-5 is sufficient. In case during next phases, Science requirements will require a lower frame error rate, this will have a major impact on the Probe design. For instance, a frame error rate of 1e-7 requires an increase of 10% of the RF output power, with a similar impact on the power subsystem (battery size). If the RF output power cannot be resized, then the G/T of the orbiter shall be improved by 0.4 dB.

- Any improvement of the Orbiter G/T could lead to major reduction of the peak power consumption. For instance, if the Orbiter G/T is increased by 1, 2, or 3 dB, the RF output power can be decreased by -20%, -37%, or -50% respectively.
- Similarly, any reduction of the bitrate (that is sized according to Assumption 4) leads to an equal percentage reduction of the RF output power. For instance, 0.5 kbps leads to -75% RF output power.

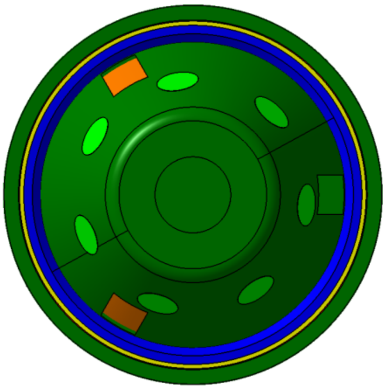

Figure 14-2: A first guess for the sizing of UHF conformal array (backshell seen from the the top)

14.1.3.1 Technical budgets

In this section a summary of the main technical budgets for the Probe communication subsystem is reported.

Table 14-2 reports the mass budget, and it can be seen that the mass of the communication subsystem has been estimated to 12.55 kg (including margins).

	mass (kg)	mass margin (%)	mass incl. margin (kg)
⊟ **Probe (Probe)**	**11.00**	**14.05**	**12.55**
⊟ **DM (Descent Module)**	**7.20**	**16.18**	**8.37**
RFDN_UHF (UHF Radio Frequency Distribution Network)	0.50	10.00	0.55
UHF_LGA_Helix (UHF Low Gain Antenna)	1.50	5.00	1.58
UHF_SSPA (UHF Solid State Power Amplifier)	0.80	20.00	0.96
UHF_SSPA_2 (UHF Solid State Power Amplifier #2)	0.80	20.00	0.96
UHF_TX (UHF Transmitter)	1.80	20.00	2.16
UHF_TX_2 (UHF Transmitter #2)	1.80	20.00	2.16
⊞ **UHF_LGA (UHF Patch LGA)**	**3.80**	**10.00**	**4.18**

Table 14-2: Probe communication subsystem mass budget

Table 14-3 shows the power budget. For the communication subsystem, the worst case peak power consumption is when one of the two transmitters is turned ON, i.e., 267 W. However, it is pointed out that this estimation was done considering a low efficiency of the SSPA, and such value should be reviewed during next phases based on BioMass heritage.

Power (W)	P_on	P_stby
⊟ Probe (Probe)	543.33	0.00
⊟ DM (Descent Module)	543.33	0.00
RFDN_UHF (UHF Radio Frequency Distribution Network)	0.00	0.00
UHF_LGA_Helix (UHF Low Gain Antenna)	0.00	0.00
UHF_SSPA (UHF Solid State Power Amplifier)	266.67	0.00
UHF_SSPA_2 (UHF Solid State Power Amplifier #2)	266.67	0.00
UHF_TX (UHF Transmitter)	5.00	0.00
UHF_TX_2 (UHF Transmitter #2)	5.00	0.00
⊞ UHF_LGA (UHF Patch LGA)	0.00	0.00

Table 14-3: Probe communication subsystem power budget

The preliminary link budget for Probe-to-Orbiter communications is shown in Table 14-1 for the UHF transmitter.

Finally, Table 14-4 shows the link budget for having carrier detection on ground by means of an array of Radio telescopes on Earth. The considered array, as reference, is the giant metrowave radio telescopes (GMRT) located in Pune (Narayangaon), India. It can be seen that, for 80 W of RF output power, an array of 20 antennas is sufficient to provide 4.6 dB of carrier-to-noise power spectral density ratio (C/N0) for carrier detection.

PARAMETER	Value	Notes
RANGE [AU]	31.4	Worst case AU
RANGE [km]	4697373149.4	
FREQUENCY [MHz]	450	
TX POWER [W]	80	
TX ANTENNA GAIN [dB]	-0.63	LGA
TX LOSSES [dB]	1	Preliminary estimation
TX EIRP [dBW]	**17.40**	**Calculated**
PATH LOSSES [dB]	**278.94**	**Calculated**
ATMOSPHERE LOSS [dB]	10.65	Best estimation
GROUND ARRAY	10.00	20 antennas, Taking into account 50% efficiency
RX G/T [dBK]	55.30	
DEMOD. LOSS [dB]	1.00	Estimation
MOD. LOSS [dB]	6.06	Suppressed carrier modulation
C/N0 [dBK]	4.64	

Table 14-4: Link budget computation for GMRT

14.1.4 List of Equipment

The transmitter considered for the baseline design is a modified version of the ExoMars UHF transceiver, limited to the transmitter module, and without the Proximity-1 data link implementation (open-loop transmission). As the technology is well consolidated, and requiring just a dedicated unit implementation, the TRL is estimated as 4, and for IceGiants an EQM approach is expected.

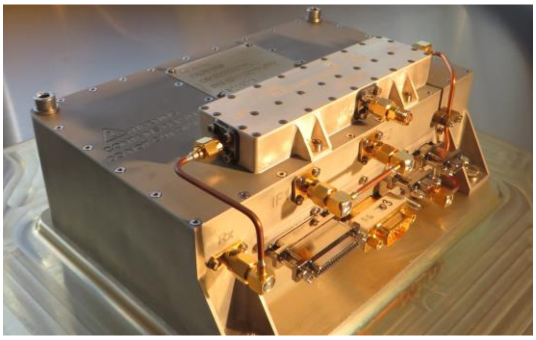

Figure 14-3: ExoMars UHF transceiver

The SSPA considered for the baseline design comes from BioMass heritage, and it is an SSPA using the high power semiconductor gallium nitride (GaN). It was implemented during the GREAT2 initiative. Such amplifier comes in two packaging solution, 15 W and 80 W (here considered). The corresponding TRL is 8. The environmental conditions of Uranus and Neptune needs to be assessed for possible delta-qualifications needed.

The backshell antenna is a conformal array technology, made of patches. The technology is well proven, TRL 5, but a delta-design shall be done specifically for IceGiant. Thus an EQM approach is expected.

Instead, for the second LGA a quadrifilar helix antenna (based on ExoMars 2016 heritage) has been considered. Such antenna can provide >0 dB between -80 and 80 degrees (as shown in Figure 14-4) from boresight, and has TRL 9 for Mars. The environmental conditions of Uranus and Neptune needs to be assessed for possible delta-qualifications needed.

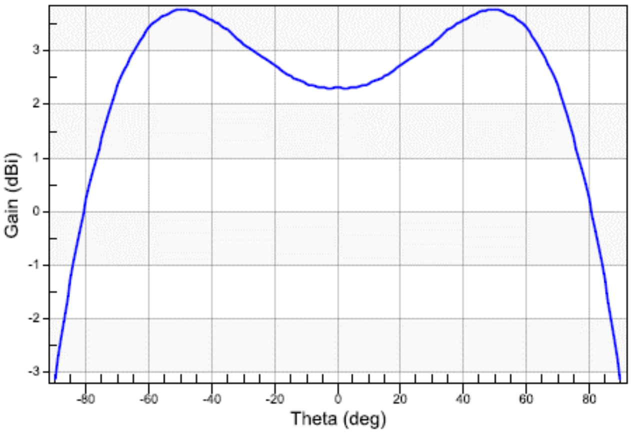

Figure 14-4: Quadrifilar helix antenna gain considered

Finally the RFDN, composed by switches and coaxial cables, still rely on ExoMars heritage, and it has TRL 9 for Mars. The environmental conditions of Uranus and Neptune needs to be assessed for possible delta-qualifications needed.

14.1.5 Options

The main option is an UHF communication subsystem with a resized RF output power. In particular, during the CDF study a bitrate of 2 kbps was assumed, that has driven the RF output power and, in turn, the size of the power subsystem.

In this respect, during Phase A it is strongly recommended to review the instrument data generation, and trade-off the number of instruments against required bitrate. Notice however, that the RF output power should not be decreased below 56W, otherwise the carrier detection by means of an array of telescopes could not be feasible (see GMRT link budget in Section 14.1.3.1).

14.1.6 Technology Needs

The following table shows the technology needs.

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
*	Baseline	UHF transmitter	QinetiQ	4		ExoMars heritage
*	Baseline	UHF antenna array on Probe backshell	TAS-I	5		Studied in the framework of ECOMTEC for EDL, ESA contract 4000113507/NL/FE

14.2 Orbiter

14.2.1 Requirements and Design Drivers

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
COM-010	The telecommunication subsystem shall be able to perform the following functions regardless of the spacecraft's attitude, throughout all the mission phases: • Receive and demodulate the uplink signal from the ground segment and transmit the telecommands (TC) to the data handling system as defined in RD[29] and RD[30]. • Receive a telemetry (TM) data stream from the data handling system and transmit this data to the ground segment as defined in RD[29] and RD[31], • Receive, transponder, and re-transmit a ranging signal as defined in RD[32], • Provide two-way Doppler.	

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
COM-020	Active (hot) redundancy shall be provided for telecommand (uplink) and passive (cold) redundancy for telemetry (downlink).	
COM-030	The link budget margins shall be as defined in RD[29]: • Nominal > 3 dB • Mean 3*sigma > 0 dB • RSS worst case > 0 dB	ECSS-E-ST-50-05C Req. 8.3.2-i
COM-040	The frequency assignment shall be done in coordination with the Space Frequency Coordination Group (SFCG) and in compliance to its recommendations and resolutions RD[33].	

The main design drivers for the communication subsystem was the *Data volume return*: the amount of data directly drives the minimum RF output power that, in turn, drives the peak power consumption.

14.2.2 Assumptions and Trade-Offs

Assumptions	
1	Cryo-cooling is adopted in the ground segment for improving G/T
2	A G/S antenna array of 2 elements (as minimum) is adopted for receiving the TM signal
3	A TM bitrate of 40 kbps is sufficient for achieving the Science requirements
4	The antenna diameters shall be less than 3 m

Concerning Assumption 1, ESA is planning an improvement for the ESTRACK network. The key item is the replacement of the feed and low noise amplifier subsystem with a single integrated subsystem including a portion of the feed and the low noise amplification, both cooled down at cryo temperature, hence called “cryo feed” or “cry cooling”. The upgrade will reduce the system noise temperature, especially acting on the lossy portion of the feed which today, due to its ambient temperature, is a major noise contributor. Quantitatively speaking, the following G/T had been assumed during the CDF study:

- G/T>52.5 dBK for a 35 m G/S in X-Band (8400-8450 MHz),
- G/T>61.3 dBK for a 35 m G/S in Ka-Band (31 800-32 300 MHz).

Concerning Assumption 2, the baseline RF link budget assumed multiple co-located ground stations in an array configuration. This situation can take place in case there are multiple 35 meters ground stations co-located in the same premises (e.g. Cebreros site or Malargue site). Then the technique relies in receiving the same signal from the multiple sites and combining it appropriately. The theoretical gain is then equal to the number of stations used for combining the signal. Two G/S lead therefore to a theoretical improvement of 3 dB in the G/T, although lower values are actually expected because of the implementation losses. In this respect, during the CDF study it has been considered that an array of 2 elements is able to provide the following G/T values:

- G/T>54 dBK in X-Band,
- G/T>63.8 dBK in Ka-Band.

Assumption 3 comes from the computation done at system level of the data volume against transmission time and ground station visibility and availability. In particular, based on the input provided by Mission Analysis, a minimum bitrate of 40 kbps had been derived for achieving the Science requirements.

Finally, assumption 4 comes from Deep Space mission heritage. Although an antenna with diameter larger than 3 m could provide better gains, thus increasing the bitrate, on the other hand it complicates the S/C accommodation, configuration, and pointing requirements. For instance:

- An antenna of 3.5 meters could provide +35% higher bitrate, but the pointing requirement would be <0.04 deg, and mass higher by +35%,
- An antenna of 4.0 meters could provide +77% higher bitrate, but the pointing requirement would be <0.03 deg, and mass higher by +77%.

Hence, during the CDF study an antenna diameter of maximum 3 meters was considered. However it is pointed out that in case during Phase A the required RF output power or bitrates could become critical, the trade-off on the antenna diameter can be re-opened.

14.2.2.1 Frequency allocation trade-off

During the CDF study a frequency allocation trade-off was done. In particular the following two options were considered:

- *X/X option*: uplink in 7145-7190 MHz, and downlink in 8400-8450 MHz,
- *X/X/Ka option*: uplink in 7145-7190 MHz, TM downlink in 8400-8450 MHz, and payload TM downlink in 31 800-32 300 MHz.

These options were considered for different RF output powers in terms of maximum bitrate, TRL, ground operations, mass, power, and cost. The trade-off is summarised in Table 14-5. It can be seen that currently the trade-off is driven by Assumption 4, i.e., the minimum TM bitrate. Hence, the only feasible option is X/X/Ka with a travelling wave tube amplifier able to provide 100 W of RF output power, although such TWTAs have TRL 2 (in the ESA member states).

ID	Allocation	Max TM bitra	TRL	G/S	Mass	Power	Cost
1	X/X, 35W	5.6 kbps	9	Cryo+Array	52-57	110-115	
2	X/X, 65W	10.5 kbps	9	Cryo+Array	52-57	165-170	
3	X/X, 80W	12.9 kbps	9	Cryo+Array	52-57	195-200	
4	X/X/Ka, 35W	14.1 kbps	9	Cryo+Array	60-65	110-115	
5	X/X/Ka, 100W	42.6 kbps	2 (TWTA)	Cryo+Array	60-65	200-220	
6	Ka/Ka, 35W	14.1 kbps	2 (XPND)	No uplink	52-57	110-115	

Legend
Best
Good
Poor
Not feasible

Table 14-5: Frequency allocation trade-off summary

14.2.3 Baseline Design

The baseline design of the X-Band communication subsystem foresees architecture as shown in Figure 14-5 and includes:

- Two X/X/Ka transponders
- Two LGAs
- One HGA
- Two Ka-Band TWTAs, and two X-Band TWTAs
- The RFDN that interconnects all the aforementioned devices.

Of the two transponders, only one is adopted for nominal operation. The second transponder is used for redundancy: its transmitter is operating in cold mode and its receiver in hot mode (see requirement COM-020).

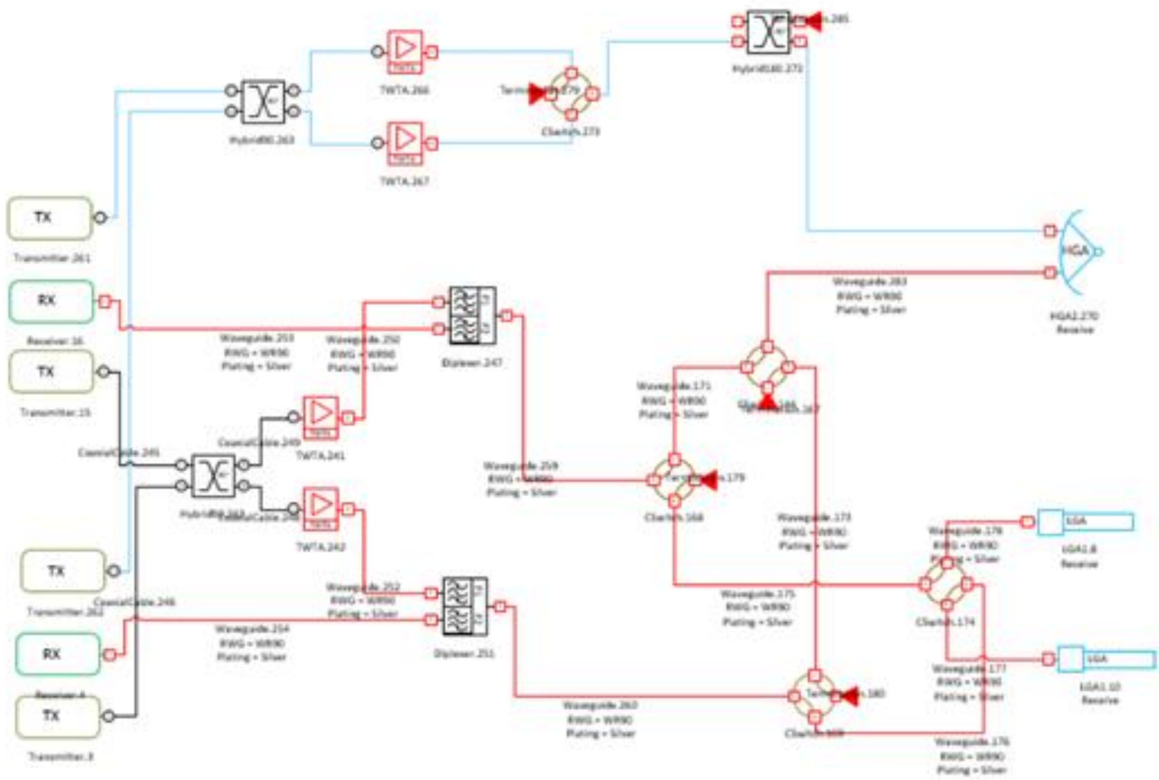

Figure 14-5: Baseline design for the Orbiter communication subsystem

The output TM signal from the active transmitter is amplified by means of TWTAs, either in X-Band for standard TT&C, or in Ka-Band for payload telemetry. The TWTA RF output power is 65 W for X-Band, and 100 W for Ka-Band.

The uplink and downlink telemetry signals are routed between the transponders and the LGAs (for low bit rate TM) or the HGA (for high bit rate TM) by means of the RFDN. The two LGAs are on opposite directions and polarizations for obtaining an almost omnidirectional coverage.

The RFDN consist of hybrids, switches, and waveguides that interconnect all the aforementioned equipment. A possible selection of the RFDN is provided in Figure 14-5, but it is pointed out that a more detailed RFDN design shall be performed during next phases by trading off reliability, dimension, mass, and power losses and its optimisation is out of the scope of the CDF study.

Finally, it is highlighted that the communication subsystem is also interfaced (by means of the RFDN) to the KaT, for supporting Radio Science with Ka-Band uplink and downlink. In particular, the KaT with diplexer shall be connected to the upper-left hybrid port shown in the upper-right corner of Figure 14-5. With this approach, the communication subsystem can support three radio science links (X/X, X/Ka, and Ka/Ka) simultaneously.

14.2.3.1 Main functions and operations

The communication subsystem is able to provide 500 bps in uplink and 1 kbps in downlink in X-Band at the farthest distance from Earth by means of the HGA. Additionally, the Ka-Band link can provide 42.5 kbps.

During safe mode it was considered that spacecraft can perform Sun-acquisition and Earth pointing of the HGA, hence still allowing 500 bps and 1 kbps in uplink and downlink respectively. In case of star-tracker failures, or other limitations, the HGA pointing to Earth can also rely on S/C strobing.

Finally it is pointed out that, differently from other missions, power flux density constraints shall not drive the RFDN design. The S/C telemetry RF link will be only activated in proximity of Neptune, while all status checks and communications before then will be by means of the umbilical link.

14.2.3.2 Technical budgets

Table 14-6 shows the mass budget for the communication subsystem. It can be seen that mass is estimated 72 kg, including margin.

	mass (kg)	mass margin (%)	mass incl. margin (kg)
SC (Spacecraft)	**64.20**	**11.59**	**71.64**
HGA (High Gain Antenna)	33.00	10.00	36.30
KaEPC (Ka-Band Electronic Power Conditioning)	1.30	20.00	1.56
KaEPC_RED (Ka-Band Electronic Power Conditioning - Redundant)	1.30	20.00	1.56
KaTWT (Ka-Band Traveling Wave Tube)	0.80	20.00	0.96
KaTWT_RED (Ka-Band Traveling Wave Tube - Redundant)	0.80	20.00	0.96
LGA_LHCP (Low Gain Antenna - LHCP)	0.90	5.00	0.95
LGA_RHCP (Low Gain Antenna - RHCP)	0.90	5.00	0.95
RFDN (Radio Frequency Distribution Network)	13.00	20.00	15.60
XEPC (X-Band Electronic Power Conditioning)	1.30	5.00	1.37
XEPC_RED (X-Band Electronic Power Conditioning - Redundant)	1.30	5.00	1.37
XKa_XPND_RED (X/X/Ka-Band Transponder - Redundant)	4.00	5.00	4.20
XKaXPND (X/X/Ka-Band Transponder)	4.00	5.00	4.20
XTWT (X-Band Traveling Wave Tube)	0.80	5.00	0.84
XTWT_RED (X-Band Traveling Wave Tube - Redundant)	0.80	5.00	0.84
Grand Total	**64.20**	**11.59**	**71.64**

Table 14-6: Mass budget for the communication subsystem

Table 14-7 shows the power budget for the communication subsystem. The worst case peak power consumption is when performing radio science, i.e., when both the X-Band and Ka-Band link are active. In such case the consumption is about ~350 W.

Power (W)	P_on	P_stby
SC (Spacecraft)	668.91	0.00
HGA (High Gain Antenna)	0.00	0.00
KaEPC (Ka-Band Electronic Power Conditioning)	9.07	0.00
KaEPC_RED (Ka-Band Electronic Power Conditioning - Redundant)	9.07	0.00
KaTWT (Ka-Band Traveling Wave Tube)	172.41	0.00
KaTWT_RED (Ka-Band Traveling Wave Tube - Redundant)	172.41	0.00
LGA_LHCP (Low Gain Antenna - LHCP)	0.00	0.00
LGA_RHCP (Low Gain Antenna - RHCP)	0.00	0.00
RFDN (Radio Frequency Distribution Network)	0.00	0.00
XEPC (X-Band Electronic Power Conditioning)	5.90	0.00
XEPC_RED (X-Band Electronic Power Conditioning - Redundant)	5.90	0.00
XKa_XPND_RED (X/X/Ka-Band Transponder - Redundant)	35.00	0.00
XPND_TX (Transponder Transmitter)	20.00	0.00
XPND_RX (Transponder Receiver)	15.00	0.00
XTWT (X-Band Traveling Wave Tube)	112.07	0.00
XTWT_RED (X-Band Traveling Wave Tube - Redundant)	112.07	0.00

Table 14-7: Power budget for the communication subsystem

14.2.4 List of Equipment

The transponder considered for the baseline design is the X/X/Ka-Band transponder of BepiColombo developed by Thales-Italy. The transponder, shown in Figure 14-6, has TRL 9 and meets all performance and functional requirements foreseen for IceGiant.

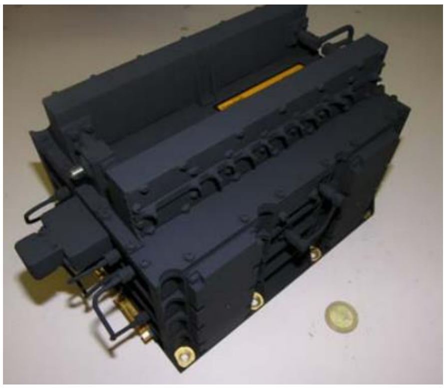

Figure 14-6: X-Band Transponder

Similarly, a possible solution for the X-Band LGAs is manufactured by TRYO and is shown in Figure 14-7. Their mass is 0.4 kg, diameter 90 mm, and height 240 mm, and they have TRL 9. Instead, the HGA of 3 meters can be a resized version of the one adopted in BepiColombo.

Figure 14-7: X-Band LGA

Concerning TWTAs for Ka-Band 32 GHz, a European technology development has been considered in the baseline design. In particular, a delta-design based on the TH4606 (shown in Figure 14-8) has been assumed, with a Breadboard+EQM+PFM+FMs approach. Hence, the TRL has been considered equal to 2.

Another possible solution for the TWTA in X-Band is the TH4704C, developed by Thales that relies on Venus and Mars Express heritage and it has TRL 9.

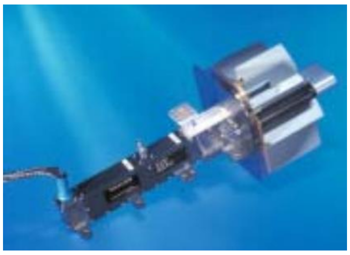

Figure 14-8: TWTA for Ka-Band 32 GHz

Finally, the RFDN elements are developed by different manufacturers, but typically a common procurement at RF harness or communication assembly level can be done. An example is TRYO procurement for hybrids, coaxial cables, and waveguides. All elements have TRL 9.

14.2.5 Options

14.2.5.1 TWTA procurement in non-ESA member states

Currently space qualified TWTAs in Ka-Band, in the ESA member states, have a maximum RF output power of 35 W. Hence, the 100 W TWTA of the baseline design implies a technology development, with a risk due to its low TRL.

As alternative option, a procurement in non-ESA member states can be done. An example, it is the TWTA manufactured by L3 in US, the 999H. According to the information public available, the TWTA could need a delta-qualification and TRL 5 is estimated.

14.2.5.2 IDST transponder

A valid option for the communication subsystem is the use of the integrated Deep-Space transponder (IDST). Recently, the IDST developed the first breadboard, reaching TRL 4, and now a technology development for an EM and mass reduction is being kicked-off.

The IDST provides several improvements with respect the BepiColombo heritage. A preliminary (but not complete) list is:

- Mass (<3.5 kg) and power reduction, by simplifying the transmitter and receiving chains
- Flexible turn-around ratio
- Implementation of LDPC in uplink, allowing 1 kbps or higher
- On-board radio science (OBRAS)
- Radio science in X/X, X/Ka, Ka/X, and Ka/Ka
- Wide-band delta-DOR
- Regenerative PN-ranging up to 25 Mcps
- Acquisition and tracking with larger Doppler values
- Autonomous receiver capabilities.

14.2.6 Technology Needs

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
*	Baseline	Ka-Band TWTA 100 W	Thales	2		
	Option	Integrated Deep Space Transponder		4		

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15 NEPTUNE DATA HANDLING

IMPORTANT NOTE: Post IFP the design of the DHS for the probe was changed. This has affected the budgets and the DH modes. The latest design is reflected in the following chapter but not flown down to the other domains or the systems chapter.

15.1 Atmospheric Probe

15.1.1 Requirements and Design Drivers

The probe DHS shall be compliant to the following requirements.

	Probe DHS Requirements
Req. ID	**Statement**
P-DHS-010	The probe DHS design shall support the Neptune & the Uranus atmosphere probing
P-DHS - 020	The probe DHS shall be off during the Orbiter launch phase
DHS -030	The probe DHS, except its timer function, shall be off during the Orbiter cruise phase, except during checkout mode
DHS -050	The probe DHS shall interface and manage the probe instruments that are: • Imaging System Instrument • In Situ Science Package Instrument • Gaz Analyser / Mass Spectrometer Instrument • Magnetometer and Plasma Monitor Instrument
P-DHS - 060	The probe DHS design shall not be redundant (TBC)
P-DHS -070	The probe DHS design budget in terms of mass and dimension shall be optimised and reduced to the needs
P-DHS - 080	The probe DHS design shall be independent to any mission configuration and it shall support 90 minutes sciences operations with a 64Mbits data volume
P-DHS - 090	The probe DHS design shall use technology with a TRL not lower than 6 by 2022
P-DHS - 0100	The probe DHS design shall support the following phases: • Cruise (including Check out mode) phase • Coasting phase • Entry & Descent phase
P-DHS - 0110	The probe DHS power consumption over all phases shall be minimised.

15.1.2 Assumptions and Trade-Offs

For the launch and cruise to Neptune or Uranus, the probe will be attached to the Orbiter. The power will be transferred from the Orbiter to the probe to keep the probe battery at full charge until separation. A hard-line TM/TC interface allows data exchange between the probe, through the orbiter, and the ground and vice versa for telecommands.

It is assumed that during the Orbiter Launch and cruise phases the probe DHS is not powered permanently, thus it will be in hibernation mode. This mode will limit the power drown from the battery.

It is also assumed that during the Orbiter launch and cruise phases that a probe checkout mode will be possible. This check out mode will allow the monitoring of the probe DHS, the instruments and other sub-systems including the TCS.

Considering requirement P-DHS -0110, the launch and cruise phase is assumed including sub-modes optimising the power consumption. Threeo sub-modes can be identified with a low power option:

- Hibernation mode: In this mode, the probe DHS processor and its peripheral components are powered on and initialised. The probe SW is loaded. No science operations are performed.
- Check-out mode: In this mode, the probe DHS is receiving TC from the Orbiter, acquiring the probe HK TMs and transmitting it back to the Orbiter.

During the Entry & Descent mode, the probe DHS is on. It is performing the data management of the probe sub-systems and instruments. In this mode, the probe DHS consumethe maximum power budget allocated to it.

It is assumed that no GNC activity is expected from the DHS probe. Moreover, considering the short descent time 90 minutes, it is assumed that not redundant design is envisaged for the probe DHS as the switching time between nominal and redundant design does not fit with the descent time.

Thus, the probe DHS shall implement the following functions:

- The computing function supporting the probe SW and related storage environment
- The communication function with the 5 probe instruments
- The mission Timer function needed to wake up the DHS at the entry time
- The probe HK acquisition function of the DHS and the probe instruments
- The probe instrument actuator commanding, if needed
- The probe IMU TM/TC handling.

For mass and power optimisation, the trade-offs are addressed through 2 criteria related respectively to the transmission function and to the mission timer function. These two criteria are:

Criteria A, which is related to the probe timer: Independent Timer vs. a Timer dependant on the CPU oscillator

Criteria B, which is related to the technology used to transfer the science data from probe instrument to the probe storage function: SpW links vs. CAN bus.

The criteria options and related pros & cons are indicated in Table 15-1.

	Option 1: Use independent Probe Timer		**Option 2: Use a timer linked to the probe CPU oscillator**		**Option 3: Use SpW communication technology**		**Option 4: Use Can bus communication technology**	
	Pros	Cons	Pros	Cons	Pros	Cons	Pros	Cons
Criterion A: Probe Mission Timer	• Independent and flexible design • Limited power during descent	• Small mass increase due to the extra timer supply	• Extra power consumption	• Mass limited to the Timer implementation only, Timer supply to be provided by - extra DC/DC during the coast and descent modes of the probe				
Criterion B: Probe DHS bus communication with the instrument					• Limited power consumption • Highly reliable communication protocol • Very flexible AIT/AIV exercice as they can be tested and integrated individually	• No miniaturised version of connectors • Difficulty to accommodate the 10 SpW cables of the instruments inside the probe	• CAN technology is flying with similar functionaity and performances.	• High power consumption per node

Table 15-1: Criteria options for DHS

For the first criteria, an activity has been performed by TERMA [RD-7]. It is allowing the provision of a Mission Timer requesting only 5mW. This option is very appealing, considering its flexibility w.r.t to the probe DHS and its very low power need. This option is the preferred one and it is the one proposed and assessed for the probe DHS.

However, at this stage of the study, the TRL achieved in this activity has not been confirmed compatible with requirement P-DHS -090.

For the second criteria, the very low power capability of the SpW transceivers is very attractive. Moreover, it allows the independent test of the transfer function of each probe instrument, when ready. This is optimum, in case of instrument delivery delays. Thus, the preferred option is to use the SpW technology for the communication function.

15.1.3 Baseline Design

For the implementation of the probe DHS functions, the following architecture, indicated by Figure 15-1 and by Figure 15-2, has been considered for the resource budget assessment:

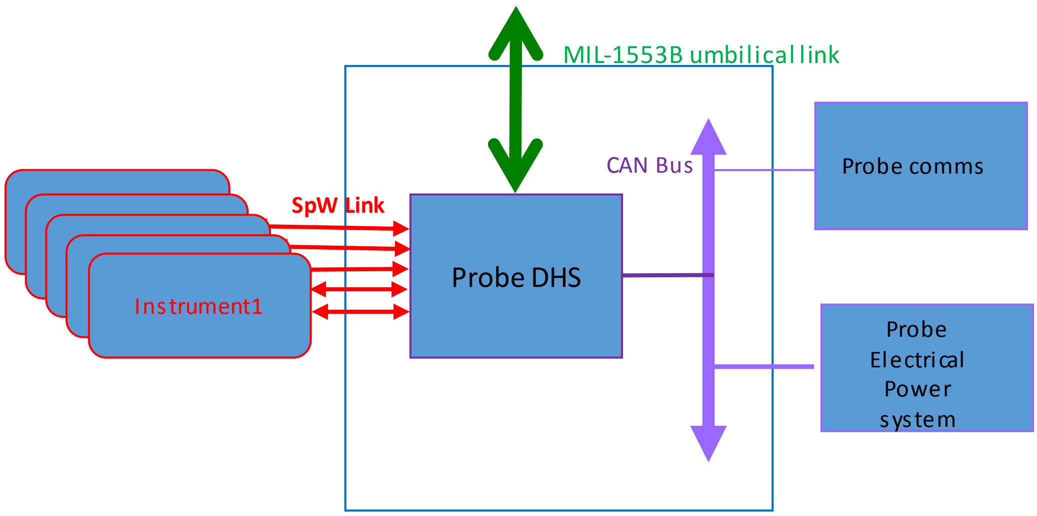

Figure 15-1: Probe DHS Baseline Architecture

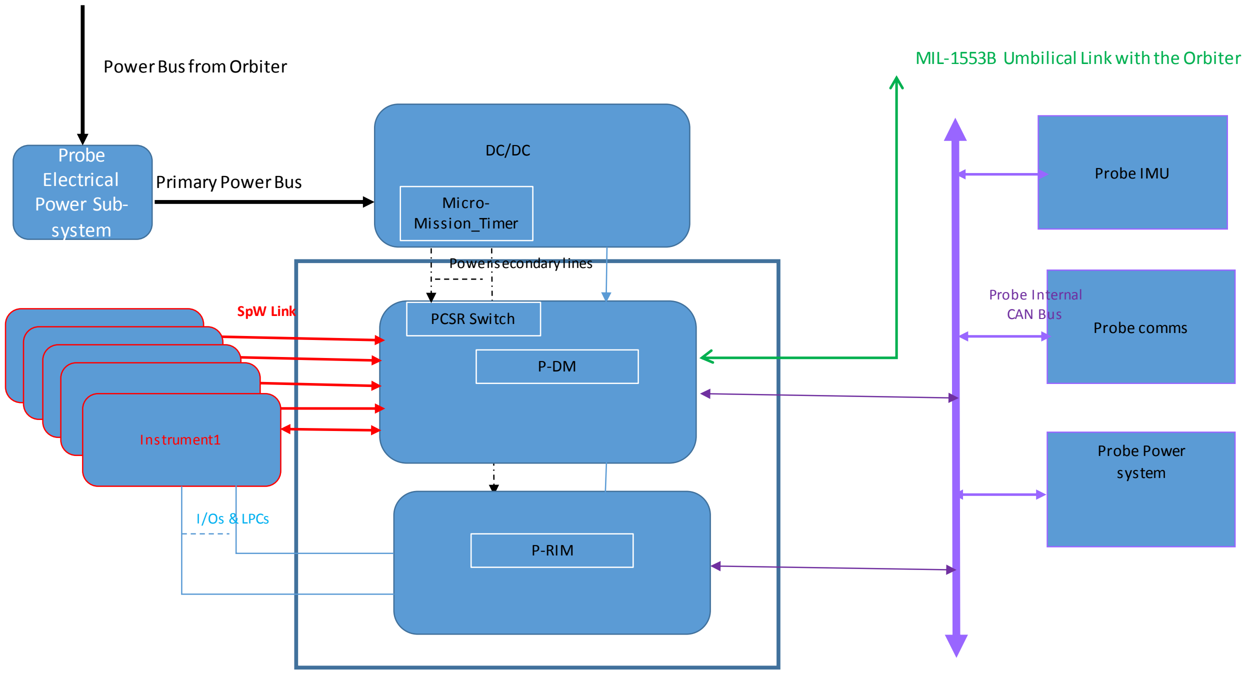

Figure 15-2: Probe DHS Detailed Architecture

This probe DHS architecture, depicted by Figure 15-2, is to be implemented on a single board. The architecture board block functions are listed here below:

- A Core Processor based on a LEON2-FT or LEON3FT processor and related resources:
  - PROM for the SW Boot
  - I/O Drivers services
  - Non-volatile and Volatile Memories
- On-board time generation and synchronisation services
- 4 SpW links I/F dedicated to the instrument operations and science data acquisition
- CAN bus dedicated to the command and control of the probe communication and power sub-system units
- RS422 for Telecommand and Telemetries interfaces with the UHF transceiver
- MIL 1553 for the Interface with the Orbiter for the monitoring during cruise phase
- Mission Micro-Timers
- DC/DC

Note: the UHF receiver is considered to be part of the probe communication sub-system.

15.1.4 Probe DHS List of Equipment

The probe DHS list of equipment is actually based on the CDMP box, based on mainly on the 3 non redundant following modules:

- Probe-DHS Module (P-DM)
- Probe Mission Micro-Timer Module (P-MμTM) based on triple Micro-Timers
- Probe Remote Interface Module (P-RIM)

These 3 modules are part of the same and single equipment;

Name	Mass (kg)	Mass Margin (%)	Mass including margin (kg)
P-DM	1	20	1.2
P-MμTM	0.02 x 3		0.024 x 3
P-RIM	0.416	20	0.5
DC/DC	0.8	20	1
Total Mass of CDMP	2.227		2.68

Table 15-2: Probe-DHS Mass Budget

Equipment Name	Power (W)	Power Margin (%)	Power Including Margin (W)
P-DM	4	20	4.8
P-MμTM			
P-RIM	0.005 x 3	20	0.006 x 3
DC/DC	5	20	6
Total Power of CDMPl	9.015		10.818

Table 15-3: Probe-DHS Power Budget

15.1.5 Technology Needs

The probe timer TRL need to be raised to TRL 6

15.2 Orbiter DHS

15.2.1 Orbiter DHS Requirements and Design Drivers

The present DHS architecture is described in the light of the Space Avionics Open Interface ARchitecture (SAVOIR) set of standards and related terminology. Thus, the main functions of the Orbiter Data Handling Subsystem (DHS) are:

- Provide the On-Board Computing capability and associated memory (OBC) for the on-board S/W to enable the Spacecraft Orbiter to function autonomously (including failure management), to respond to TC and to generate TM
- Interface with science Payload equipment to collect time-stamped and format science data for downlinking them to ground Earth stations
- Provide long-term storage, in particular to store science data during long period outage
- Interface with the Orbiter platform equipment, distributing commands and collecting telemetry, formatting low-level data (e.g. AOCS equipment and thermistor acquisitions)

- Interface with the Communications subsystem to communicate with ground Earth stations.

Data Handling Sub-System Requirements	
Req. ID	**Statement**
DHS-010	The Orbiter DHS shall accommodate the mission to Neptune and to Uranus
DHS-020	The Orbiter DHS shall accommodate operations during the following operational phases: • Pre-launch Phase • LEOP Phase • Transfer Phase • Commissioning Phase • Nominal Science Operations Phase
DHS -030	For configuration & data handling aspects, the DHS shall be able to support at least the following modes: • Pre-Launch Modes for the ground test configuration and operations • Operational Mode ensuring the generation of mission products • Safe Mode ensuring safety of all spacecraft subsystems and payloads.
DHS -030	The DHS sub-system shall be compliant to the Avionics System Reference Architecture (ASRA) SAVOIR specifications related to on-board OBC, MM and RTU units as specified respectively in SAVOIR-GS-001, SAVOIR-GS-002, SAVOIR-GS-003 and SAVOIR-GS-004
DHS -040	The DHS sub-system shall ensure the spacecraft safety and be compatible with the ground outage durations
DHS -050	The Orbiter Data Storage function has to be tolerant to one ground station failure
DHS -060	The DHS sub-system shall support On-board Data Storage function which copes with a data volume up to 30 Gbits, covering 50 days of outage
DHS -070	The DHS sub-system shall support CFDP file management
DHS -080	The DHS sub-system memory at EoL shall be sufficient to store all on-board HK and science data. The science phase duration shall be of 2 years at least.
DHS -090	All DHS digital electronics and Mass Memory shall be immune from destructive Single Event Effect (e.g. Single Event Latch-up (SEL)) and protected against Single Event Effects (e.g. SEU and LET) by parts selection and circuit design.
DHS-0100	The orbiter DHS shall cope with the worst case radiations conditions related to Uranus (due to Jupiter fly-by TID) for which sensitive units shall be provided even with 10mm of Al shielding (TBC by selected Jupiter fly-by dates)
DHS -0110	The protection of on-board memory shall ensure that no mission outage occurs throughout S/C lifetime caused by SEU or SET.
DHS -0120	The orbiter DHS hardware shall be redundant
DHS -0130	The orbiter DHS shall accommodate 7 instruments, that are:

	• Camera High resolution • Magnetometer • Imaging Spectrometer* • Ion and Neutral Mass Spectrometer • Microwave radiometer • Radio Science: USO (Ultra-Stable Oscillator) • Radio Science: Ka-band Transponder
DHS-0150	The orbiter DHS shall cope with dual launch configuration that are: • stacked spacecraft configuration • - side by side spacecraft configuration
DHS -0160	The Orbiter DHS shall use technology with a TRL not lower than 6 by 2022
DHS -0170	The Orbiter DHS shall be compatible with a launch as early as 2031

15.2.2 Assumptions

The DHS assumptions made in this study are, as follow:

1. Assumptions for the CPU of the Orbiter DHS

The Computing Micro-Processor has to fulfil performances as defined by the Central SW (CSW) applications and have their characteristics driven by the technology used to manufacture them. It drives the design and the development of other electronics parts of the computer (memories, internal/external interfaces, etc). Indeed, it influences the SW development process and hence has a significant impact on the mission development. The main assumption for the CPU of the Orbiter DHS is that it is solely dedicated to CSW. Thus, the instrument data processing is performed individually by each instrument, where needed.

2. Assumptions for the Science Data Volume of the Orbiter

Following the instruments studied and presented during the CDF sessions, it becomes clear that the maximum science data volume to be stored, between consecutive ground stations, will be in the range of 30 Gbits during 30 days of science operations.

As the trend, these days, is leaning towards the usage of external memory with faster serial interfaces. Flash memory is the baseline for the implementation of the On-Board Mass Memory. There are two main types of flash memory where code and data is stored, and they are NAND flash memory and NOR flash memory. NAND has much more capacity and a higher density than NOR. Moreover, NAND Flash memory devices are commonly available in larger capacities at generally above 1 Gbits; while NOR flash memory's top capacity is around 1 Gbits.

The Flash storage space shall be organised, at partition level, to cope with redundancy aspects and to ensure the 30 Gbits End of life capacity.

Considering, present implementation scheme, a storage space up 128 Gbits should cope with the Orbiter need in terms of code and data storage.

3. Assumptions for the DHS Orbiter Mass Memory Unit

Considering the limited storage space needed by this mission, it is assumed to have a unique equipment to handle the OBC and the MMU functionalities and performances of the Orbiter DHS. This equipment is called CDMU. Actually, this design and manufacturing option will not be a premiere as it has already been used in Euclid mission.

4. Assumptions for the DHS Orbiter communication technology
The important factors of avionics buses include:

- Deterministic behaviour
- Fault tolerance
- Redundancy.

Moreover, most avionics buses are serial and multi-drop. Very often MIL-STD-1553 (rev B) is widely used in ESA satellites. Because of its success in automotive industry, CAN bus technology attracted the attention of the space industry as well. Thus today, the DHS communication bus trend seems to be moving to CAN bus. DHS will still have to accommodate few RS-422 interfaces where platform equipment require it.

This study was taking legacy from instrument designs already flying in other missions. In all the cases, SpW technology was used to interface these instruments. Thus, it is assumed that SpW technology will be used to interface the Orbiter instruments.

5. For the side-to-side configuration of ESA & NASA spacecraft, MIL-1553B communication is considered instead of the SpW solution proposed by NASA. The rational resides in two facts that are:
   - The SpW technology suffers lack of galvanic isolation capability
   - The SpW technology has a very limited common mode i.e. 1.125V with ± 1 V difference between the grounds of the 2 side-by-side satellites.

An umbilical link, Mil-1553B based, has already been successfully used in Bepi Colombo mission. Its design, with TRL 9, is assumed to be re-used for the Orbiter communication between the 2 side-by-side satellites.

15.2.3 Trade-Offs

The OBC is very central in the orbiter platform, and therefore has the potential to integrate more functions as the science data storage. As the volume of science data, to be stored on board is not huge, it is expected that the related board can be housed in the Orbiter OBC. This assumption has already been stated. The resulting CDMU will bring saving not only at mass, power and external interfaces but at manufacturing level as well. The CDMU design will not be further traded-off. The CDMU unit will have to be complemented by the usage of CFDP for the related file system management. Today, it is becoming a baseline function to collect science data and to downlink them to ground station.

The remaining Orbiter DHS functionalities resides with the Orbiter HK data collection and actuator commanding that is classically performed by a centralised RIU unit. If requested by the overall system budget, further saving in terms of mass, power, external interfaces, test effort, manufacturing time and costs, it will be interesting to consider a de-centralised architecture to implement the RIU functionalities.

15.2.4 Baseline Design

Classically, the DHS is subdivided into 3 equipment interlinked, that are:

- On-board Computer (OBC) to host S/W and associated memory, to interface the communication subsystem, to handle reconfiguration, to host on-board time function.
- Mass Memory Unit (MMU) to store science and HK data; interfaces to the Payload equipment to allow efficient storage of science data without interrupting S/W processing.
- Remote Interface Unit (RIU) to collect HK from low-level interfaces (including analogue interfaces e.g. thermistors) of many platform equipment, to command actuators.

Considering the previous assumptions and trade-offs, the baseline design of the Orbiter DHS is based on 2 units that are the CDMU and the RIU. Both of these units shall be doubled to ensure the reliability figures needed by ESA science mission. The baseline design will rely on cold redundant configuration of these units. Thus, any combination of the nominal or the redundant CDMU with the nominal or the redundant shall be possible. These two units shall be intrinsically single point failure free. They shall provide high reliability and availability during all phases of the mission. Their design shall be based on rad-hard components and possibly will need extra shielding. The extra shielding is mainly to be considered for the Neptune Orbiter DHS, considering the specific Neptune harsh radiation environment.

Note that the DHS provides the computing platform for the Central SW (CSW) but does not include the CSW itself. Thus, the CDMU should provide the adequate computing capability which is evaluated around 80 MIPS. Thus, the CDMU baseline design is based on a one core CPU device; as for example the Leon2FT (~80 MIPS @ 100Mhz).

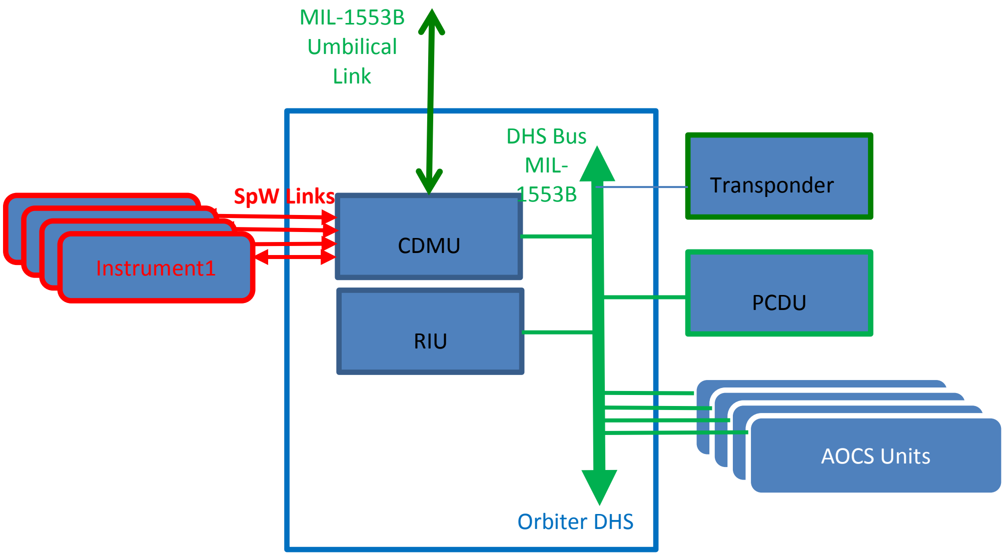

Figure 15-3: DHS Orbiter Baseline Design

15.2.5 Orbiter DHS List of Equipment

The Orbiter DHS list of equipment is based on two cold redundant units that are:

- Command & Data management Unit (CDMU)
- Remote Interface Unit (RTUC).

Equipment Name	Mass (kg)	Mass Margin (%)	Mass Including Margin (kg)
CDMU x 2	6.66 x2	20	8 x2
RTUC	13.33 x2	20	16 x2

Table 15-4: DHS Mass Budget

Equipment Name	Power (W)	Power Margin (%)	Power Including Margin
CDMU	29.15	20	35
RTUC	13.33	20	16

Table 15-5: DHS Power Budget

Equipment Name	Height (mm)	Length (mm)	Width (mm)
CDMU	200	250	300
RTUC	200	250	300

Table 15-6: DHS Volume Budget

15.2.6 Orbiter DHS Options

There are mainly 2 design options for the Orbiter DHS that are related to the RIU implementation

Option1 : DHS based on a centralised RIU (RIUC)

In this case the DHS will be based on 2 equipments that are: CDMU + RIU.

The CDMU will be based on classical OBC manufacturing with an extra board for the Mass Memory. Specific power on of the memory banks/partitions will ensure redundancy and end of life performances.

The RIU will be based on legacy design of RIU flying on ESA science mission but tailored to the need of the Orbiter in terms of monitoring and command and control of the Orbiter sensor & actuators.

Option 2: DHS based on a de-centralised RIU (RIUD)

In this case the DHS will be based on one main equipment that is: CDMS

The CDMS will be based on classical OBC manufacturing with an extra board for the Mass Memory as in option1..

The option2, via the RIUD, offers the possibility to decrease the mass by ~1 kg and the power by ~4W but this option is not yet flying. Moreover, there is no guarantee that it will be compliant to DHS -0170l.

Therefore, the RIU will be based on legacy design of RIUC flying on ESA science mission but tailored to the need of the Orbiter in terms of monitoring and command and control of the Orbiter sensor & actuators

15.2.7 Orbiter DHS Technology Needs

Similar CDMU and RIU have been either already manufactured (e.g. Euclid CDMU) or already been flown (e.g. Bepi Colombo RIU). Possibly extra shielding or minor adaption to fit to all the DHS performances might be needed but no specific technology need to be developed for both Orbiter.

16 NEPTUNE THERMAL

16.1 Atmospheric Probe

16.1.1 Requirements and Design Drivers

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
THE-010	The heatshield shall protect the inner capsule from the harsh entry environment.	
THE-020	The P/L compartment shall be maintained within [0/30°C] (TBC) during all mission phases.	
THE-030	The probe shall be able to survive a coasting phase of 20 days (TBC).	

16.1.2 Assumptions and Trade-Offs

16.1.2.1 Assumptions

Assumptions	
1	No heater power available for boost heating before probe release
2	Steep entry (FPA=-35 deg) to allow observation from Earth (resulting from SYS-060)
3	Test facilities limited to maximum 70 MW/m^2 (combined convection and radiation) within budget of M-class mission. No facility is readily available today. (see chapter 16.1.8)

16.1.2.2 Frontshield TPS Material

Dense Carbon-Phenolic

Based on the extremely harsh aerothermodynamic entry environment, in terms of atmospheric composition, peak heat fluxes and pressure loads, the only material type with relevant heritage is fully-dense carbon-phenolic. Heritage stems from the Galileo probe which entered into Jupiter's atmosphere back in 1985 and which used an American fully-dense carbon-phenolic ablator (rho = 1450 kg/m^3) developed in the 1970's. While the composition of Jupiter's atmosphere is similar to Uranus and Neptune, entry loads were more severe reaching peak heat fluxes in the order of 350 MW/m^2 (combined convective and radiative) RD[41]. Further heritage comes from the Pioneer-Venus entry probes which also used fully-dense carbon-phenolic. However, the material used for Jupiter-Galileo and Pioneer-Venus seems not to be available any more.

In Europe different types of dense carbon-phenolic materials are available. Dense carbon-phenolic materials available at ArianeGroup can be classified in two types, 2D-CP and 3D-CP. Both materials are produced at high yearly production rates.

2D dense carbon-phenolic:

- Produced through a tape wrapping and chop molding manufacturing process;
- Used as insulating layers inside nozzles of Ariane-5 & -6 solid rocket motors;
- Manufacturing well mastered for large cones up to 3m diameter allowing to produce a heatshield in one piece;

3D dense carbon-phenolic:

- 3D fiber architecture produced through a needling process;
- Improved mechanical resistance with comparable thermal performance;
- Used for insulating layers inside nozzles of Vega P80 solid rocket motors;
- Manufacturing done for large cones up to 2.1 m diameter;

For both types of the material, the performance under Uranus/Neptune entry conditions would have to be verified. However, due to similarity to the material used on Jupiter Galileo & Pioneer Venus there is confidence that the material is suited for Uranus/Neptune entry conditions. Further, the capability to produce a monolithic shield with spherical nose would have to be verified but is not expected to be a potential show-stopper.

It is highlighted that the dense carbon-phenolic materials from ArianeGroup are being considered by JPL as candidate material for the Mars Sample Return Earth Entry Vehicle (MSR-EEV).

Ceramic Materials

Alternatively also carbon-carbon (C-C) or possibly Carbon-SiC (C/SiC) materials could represent a suitable choice for the frontshield TPS. They typically demonstrate a better ablation behavior with lower recession rates than classical ablators. Mechanical properties of C-C are similar to those of Aluminum while it can be operated up to temperatures of several thousand degrees. Ceramic materials might therefore combine the functions of the TPS with those of a hot primary structure.

Manufacturing capabilities for C-C and C/SiC materials are available in Europe. Intensive development, characterization and testing was done for both, launcher and Earth entry applications. However, the performance under Uranus/Neptune entry conditions would have to be investigated.

Advanced Materials

The TPS mass fraction of an entry probe is strongly correlated to the integrated heat load until shield separation. This is demonstrated in Figure 16-1. From this, the TPS mass fraction for an Ice Giant entry probe when using classical ablators can be expected to be in the order of 30-40%.

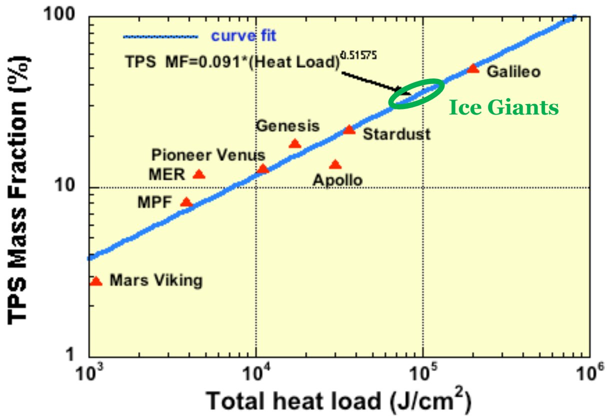

Figure 16-1: Probe TPS mass fraction over total heat load RD[41]

A considerable reduction of the TPS mass fraction requires the development of advanced TPS materials. Such development is currently performed by NASA in the frame of the 'Heatshield for Extreme Entry Environment Technology' (HEEET) program RD[41] & RD[42]. The development is based on a 3D-woven material tailored in such a way that the external (dense) part provides high resistance against recession while the internal (less dense) part provides improved thermal insulation. According to RD[42], TRL-6 is expected to be reached within 2019. Further development might be required for a specific application on an Ice Giant probe.

Frontshield Material Selection

Within this study, European fully-dense 3-D carbon phenolic has been assumed as baseline frontshield TPS material, based on the extensive existing European manufacturing capabilities with regular production of large complex shapes and the expected adequate performance based on similarity to the material used on the Galileo probe.

16.1.2.3 Backcover TPS Material

The backcover TPS is assumed to be based on the European ASTERM material developed by ArianeGroup, which is a lightweight carbon-phenolic material. ASTERM is produced with standard European raw materials based on a robust manufacturing process by impregnating a rigid graphite substrate with phenolic resin. Within the study the nominal density of 280 kg/m^3 was assumed, while adaptation is possible within a significant range to adapt to the mission needs. ASTERM has been pre-qualified (TRL-6) for an application as frontshield material of the Earth Return Capsule (ERC) of sample return missions with typical peak heat fluxes of about 14 MW/m^2, integrated heat loads of around 240 MJ/m^2, and peak stagnation pressure loads of around 1 bar.

While the material performance in an H_2/He-atmosphere would have to be verified, the expected backcover peak heat fluxes (<2 MW/m^2) and integrated heat loads (~20 MJ/m^2) are well within the qualified range of the material.

16.1.2.4 TPS Margin Approach

As described in chapter 17, for the sizing of the TPS ablator a margin of 100% has been applied on top of the calculated convective and radiative stagnation point heat fluxes.

Heat flux blocking effects due to injection of pyrolysis gas into the boundary layer have been considered with a reduction of 20% applied on both, convective and radiative stagnation point heat fluxes. Stagnation point heat fluxes have been assumed applicable over the entire frontshield.

Again as described in chapter 17, for the backcover ablator sizing 2.5% of the margined convective stagnation point flux plus 1.0% of the margined radiative stagnation point flux have been considered.

On top of the derived minimum required ablator thickness a margin of 50% has been applied, in order to reflect the unknown material behaviour under Uranus or Neptune entry conditions with extreme heat flux and pressure conditions. Finally, an additional 20% maturity margin has been applied on the derived ablator mass.

16.1.2.5 Aerothermal Heat Fluxes

The following case had been defined as baseline for the TPS sizing at entry interface:

- Relative entry velocity: 23.1 km/s
- Relative entry flight path angle: -35 deg
- Probe mass at entry: 341 kg
- Probe diameter: 1.35 m

Unfortunately, as described in chapter 17, some parameters in the ATD tool were initially set such that the considered entry velocity was assumed as inertial velocity rather than relative velocity. Therefore the initially provided ATD data was based on an inertial velocity of 23.1 km/s which corresponds to a relative velocity of only 20.6 km/s.

Since this error was only found after the design freeze, the ablator sizing was done for both cases. Figure 16-2 below provides the heatflux timelines based on an inertial velocity of 23.1 km/s as used for the baseline TPS sizing. Figure 16-3 provides the corrected heat flux timelines for a relative entry velocity of 23.1 km/s. All heat fluxes consider the margin approach as described in chapter 16.1.2.4.

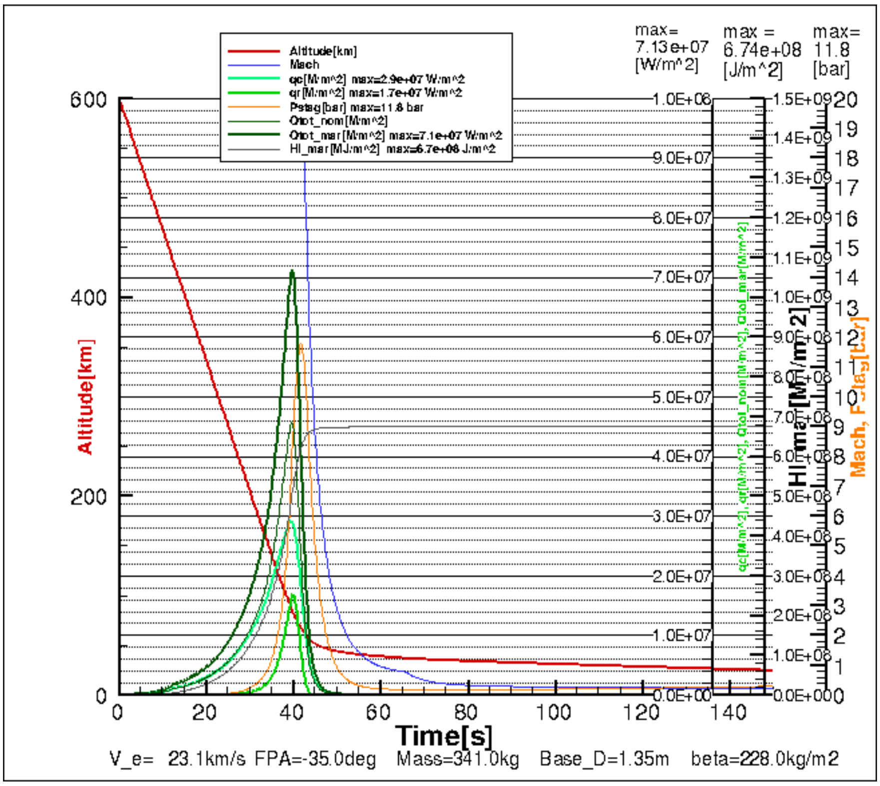

Figure 16-2: Heat flux timelines for baseline TPS sizing (20.6 km/s relative)

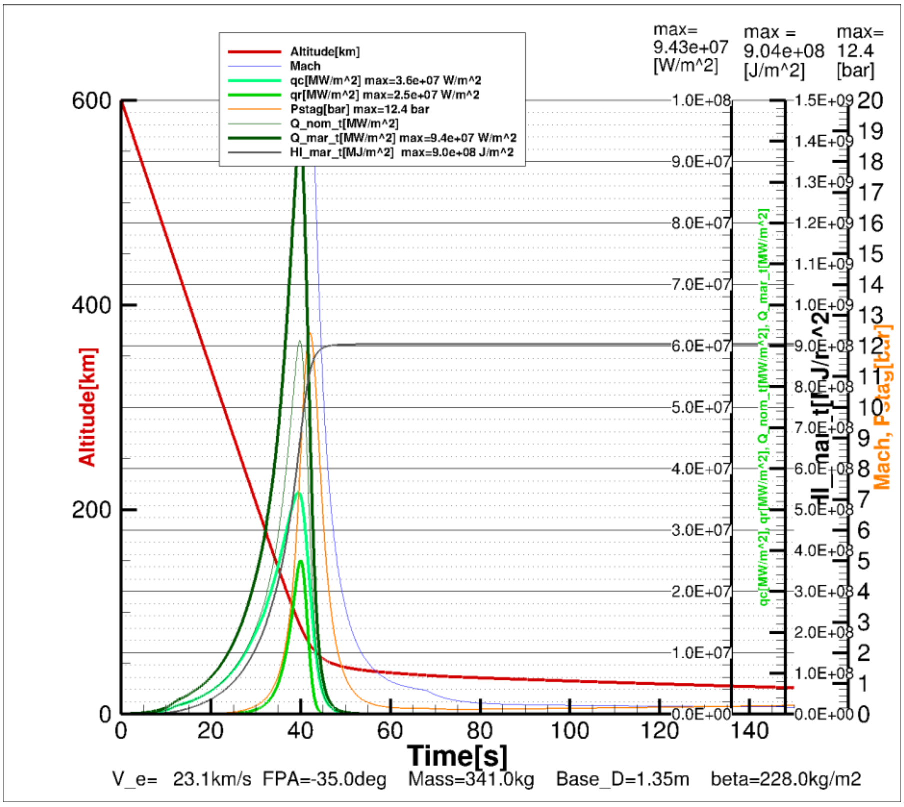

Figure 16-3: Heat flux timelines for corrected TPS sizing (23.1 km/s relative)

As can be seen, after the correction from inertial to relative (delta-v = 2.7 km/s), the peak heat flux increased from 71 MW/m² to 94 MW/m², while the integrated heat loads increased from 674 MJ/m² to 904 MJ/m².

While TPS sizing for both cases is provided in chapter 16.1.3.1, it has to be highlighted that the peak heat flux of the corrected case is beyond the test facility capabilities which are considered achievable within the budget of an M-class mission (see also 16.1.8). Further, even though no clear limitation for the fully-dense carbon-phenolic material can be stated, the high heat fluxes may also represent an increased risk for the material performance.

In conclusion: a steep entry trajectory (FPA = -35 deg combined with a relative entry velocity of 23.1 km/s) may therefore likely not be feasible. More shallow approaches should therefore be addressed in future studies.

Figure 16-4 shows the heat flux timelines used for the backcover ablator sizing, based on the assumptions described in chapter 16.1.2.4.

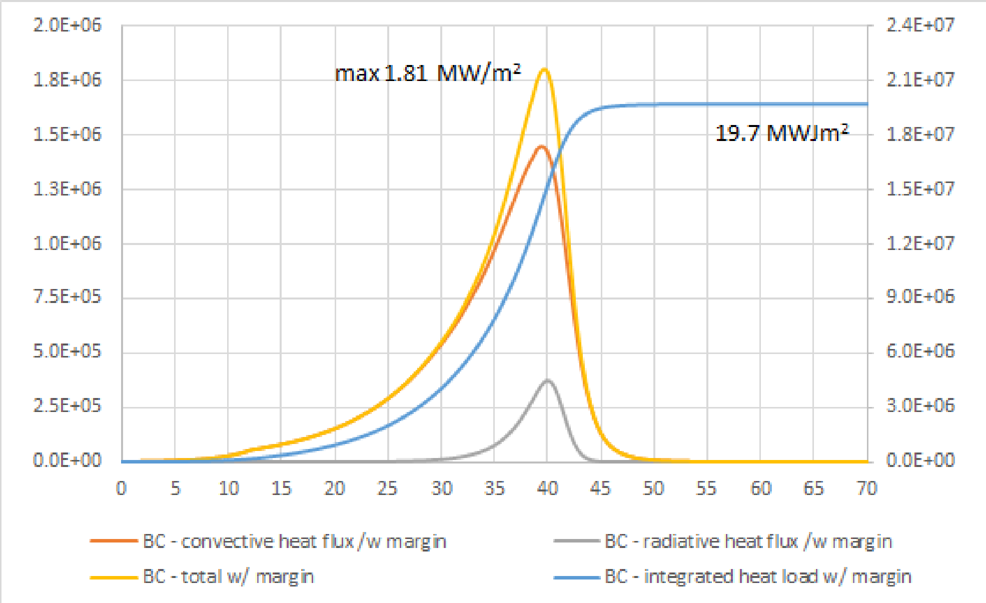

Figure 16-4: Backcover heat flux timelines

16.1.2.6 Release Sequence

The following release / timing sequence has been assumed for the heatshield sizing.

- Drogue parachute opens at Ma=0.8
- Backcover release at Ma=0.8 plus 2 seconds
- Frontshield release at Ma=0.8 plus 10 seconds

Note that the baseline frontshield release timeline assumed in the EDS sizing was Ma=0.8 plus 15 seconds (see Section 18.3.3). The effects of this should however be limited, and should be addressed in future work.

16.1.3 Baseline TPS Design

16.1.3.1 Frontshield Design

A hybrid hot structure concept has been assumed as baseline for the frontshield, i.e. a fully-dense carbon-phenolic ablator is mounted onto a ceramic CMC structure. This way the temperature limit at the bonding interface of the ablator is significantly increased, which in consequence leads to a relevant reduction of the required ablator thickness. A lightweight efficient insulation is then used to insulate the descent compartment against the hot structure. A further advantage is increased robustness e.g. against micrometeoroid impacts. High temperature structural stand-offs are required for the load-carrying mechanical connection between the hot structure and the descent compartment. The stand-offs are covered under the structure (8.1.3). Such design is also referred to as so-called SEPCORE® concept RD[44].

Within the current study the temperature limit at the ablator bonding interface has been assumed at 800°C, whereas for a 'classical' substructure in CFRP and/or Aluminum a limit of about 180°C would apply. The limit value was chosen based on work performed within the FP7 HYDRA activity performed under funding of the European Commission on adhesive based joining technologies RD[45]. It shall be noted that RD[45] also provides study results for advanced bonding techniques which indicate limits of about 1200°C and above. However, verification would be required for the material combination of interest here (fully-dense carbon-phenolic onto CMC).

The ablator sizing analysis within this study has been based on material properties from the FM5055 material for which a good agreement has been found with characterization performed on the 2D-CP from ArianeGroup. Based on indications from ArianeGroup, the performance of the 3D-CP is preliminarily assumed to be comparable.

The initial heatshield temperature at entry point has been conservatively assumed as 20°C.

Figure 16-5 shows the ablator sizing for a relative entry velocity of 20.6 km/s. To maintain the backface temperature limit of 800°C, an ablator thickness of 26.3 mm is required. Applying the 50% sizing margin, this results in a design thickness of 39.5 mm.

Figure 16-6 shows the ablator sizing for a relative entry velocity of 23.1 km/s. To maintain the backface temperature limit of 800°C, an ablator thickness of 28 mm is required. Applying the 50% sizing margin, this results in a design thickness of 42 mm.

(Note that for a frontshield release at Ma=0.8 plus 15 seconds, as assumed in the EDS sizing case, an additional ~1.5 mm thickness would be required, including margin)

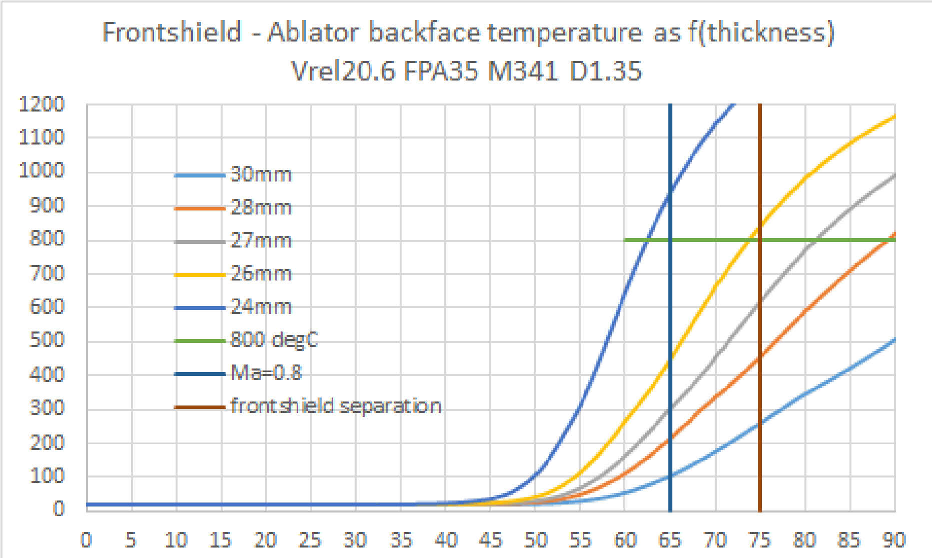

Figure 16-5: Frontshield Ablator Sizing for 20.6 km/s relative entry velocity (erroneous baseline at design freeze)

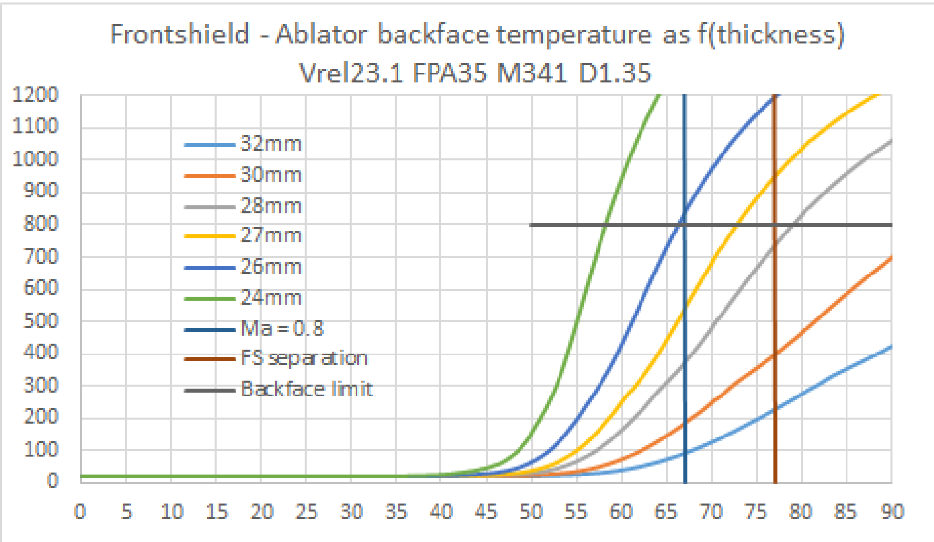

Figure 16-6: Frontshield Ablator Sizing for 23.1 km/s relative entry velocity (correction after design freeze)

16.1.3.2 Backcover Design

Due to time constraints during the study, the backcover heatshield design had initially been considered identical to the one foreseen during the PEP study in 2010, which was based on the ASTERM ablator (9mm thickness) mounted onto a hot structure. Note that in the PEP study the backcover was assumed to remain attached during the atmospheric descent for a duration of about 60 minutes.

Since the backcover is now assumed to be released together with the drogue parachute just seconds after peak heating, the thickness of the ASTERM ablator can be reduced. As shown Figure 16-7, only 4mm ablator thickness are needed to maintain the backface temperature limit of 800°C. Applying the 50% sizing margin, this results in a design thickness of 6 mm.

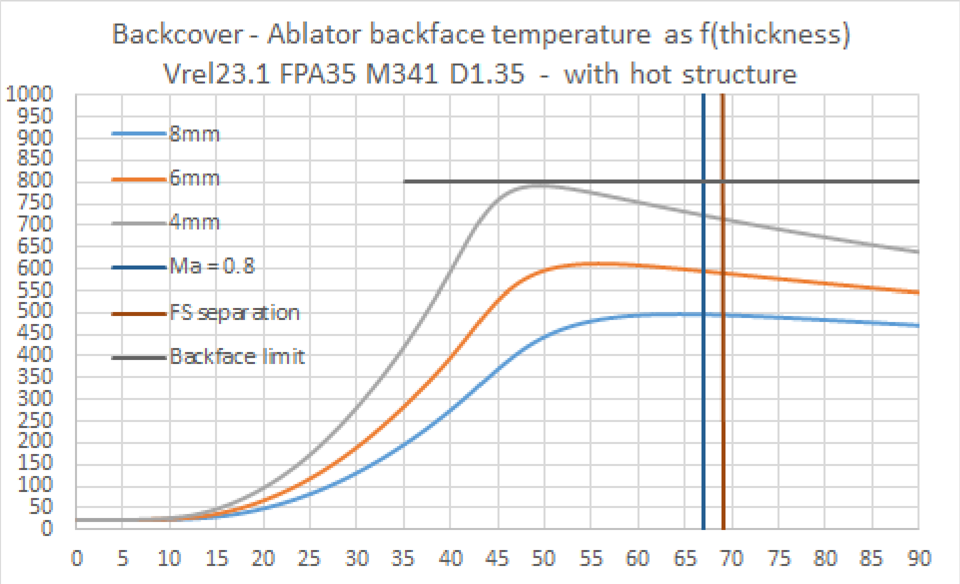

Figure 16-7: Backcover ablator sizing (update after design freeze)

16.1.4 Baseline TCS Design

The thermal control design of the probe assumes the frontshield and backcover to be externally covered by a high-performance 22-layer MLI (based on JUICE MLI with HELPAC spacer and incorporating micro-meteoroid shield capability).

31 Radioisotope Heater Units (RHU's) are used for heater power during the 20 day coasting phase. A white-painted radiator window (0.15 m²) is implemented on the frontshield to reduce the sensitivity of the internal temperature towards uncertainties in the knowledge of the MLI performance and to accounts for heater power variations due to RHU decay. This design is similar to the one applied on the Huygens probe, see Figure 16-8.

Figure 16-8: Radiative window on Huygens probe

A simplified thermal mathematical model was used to derive the number of required RHU units and to derive the required area of the radiative window, see Figure 16-9.

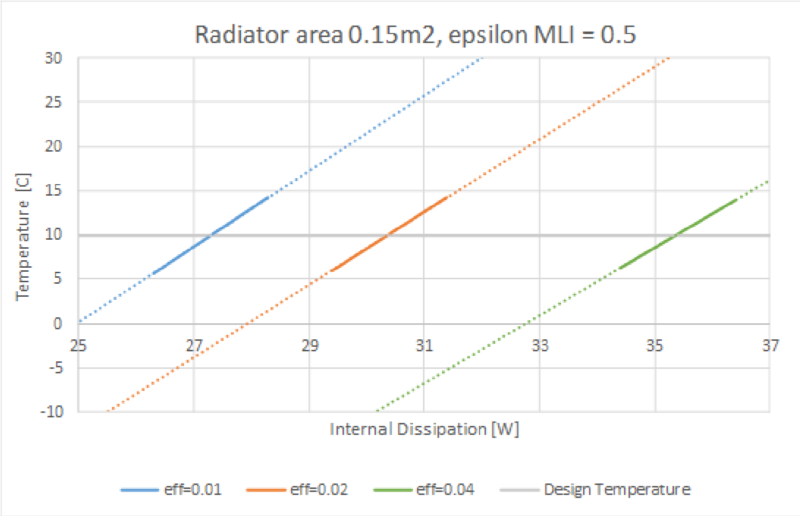

Figure 16-9: P/L temperature as function of RHU dissipation and MLI performance

Aerogel or foam insulation is used inside the pressure vessel to insulate the payload compartment during the atmospheric entry under parachute.

16.1.5 List of Equipment

16.1.5.1 Heatshield Equipment

As explained in chapter 16.1.2.5, a correction to the aerothermodynamic heat fluxes has been received after study design freeze. Further to this, after study design freeze it was also found that the surface area assumed for the frontshield was incorrect.

Therefore two versions of the heatshield equipment summary table are provided. Table 16-1 provides the heatshield equipment summary at the time of study design freeze. Table 16-2 provides the update after study design freeze reflecting 1/ the increase in frontshield ablator thickness due to the higher relative velocity, 2/ the corrected frontshield surface area, and 3/ the revised backcover design eliminating the hot structure.

	density [kg/m3]	area [m2]	thickness [mm]	Mass excl. margin [kg]	Margin		Mass incl. margin [kg]
Frontshield ablator (3D carbon-phenolic)	1350	1.82	39.5	97.05	20	%	116.46
Frontshield hot structure (C/SiC)	1800	1.82	2.4	7.86	20	%	9.43
Frontshield internal insulation	140	1.82	10.0	2.55	20	%	3.06
Backcover ablator (ASTERM)	280	1.72	9.0	4.33	20	%	5.20
Backcover hot structure	1800	1.72	2.4	7.43	20	%	8.92
Backcover internal insulation	140	1.72	20.0	4.82	20	%	5.78
Heatshield instrumentation				2.00	20	%	2.40
Subsystem total				**126.04**			**151.25**

Table 16-1: Heatshield equipment and mass (at study design freeze)

	density [kg/m3]	area [m2]	thickness [mm]	Mass excl. margin [kg]	Margin		Mass incl. margin [kg]
Frontshield ablator (3D carbon-phenolic)	1350	1.99	42.0	112.83	20	%	135.40
Frontshield hot structure (C/SiC)	1800	1.99	2.4	8.60	20	%	10.32
Frontshield internal insulation	140	1.99	10.0	2.79	20	%	3.34
Backcover ablator (ASTERM)	280	1.73	6.0	2.90	20	%	3.48
Backcover hot structure	1800	1.73	2.4	7.46	20	%	8.95
Backcover internal insulation	140	1.73	20.0	4.83	20	%	5.80
Heatshield instrumentation				2.00	20	%	2.40
Subsystem total				**141.40**			**169.69**

Table 16-2: Heatshield equipment and mass (correction after study design freeze)

16.1.5.2 TCS Equipment

The summary and masses of the thermal control equipment is provided in Table 16-3.

	density		area [m2]	quantity	Mass excl. margin [kg]	Margin		Mass incl. margin [kg]
Frontshield MLI (20-layer, incl. overlaps)	0.8	kg/m2	1.77		1.42	10	%	1.56
Frontshield radiator (white painted Alu-foil)	1	kg/m2	0.15		0.15	10	%	0.17
Backcover MLI (20-layer, incl. overlaps)	0.8	kg/m2	1.92		1.54	10	%	1.69
Pressure vessel internal insulation (Aerogel)					5.00	20	%	6.00
RHU for the PL in the probe	0.04	kg/unit		31	1.24	10	%	1.364
RHU support structure	0.1	kg/unit		31	3.10	10	%	3.41
Subsystem total					**12.44**			**14.19**

Table 16-3: TCS equipment and mass

16.1.6 Options

The following options are identified which would have to be addressed in further detail in a separate study.

1. Shallower entry flight path angle: As discussed in chapter 16.1.2.5, the steep entry flight path angle (FPA=-35deg) assumed within this study leads to very high heat fluxes which cannot be reproduced with available facilities and for which facility adaptation is likely not achievable within a Type-M mission budget. A potential delta-study should therefore assess mission concepts with a shallower entry to reduce the peak heat fluxes. Due to the increased integrated heat loads this will however require increased ablator thickness.

2. Frontshield without hot structure: Due to the very short heat flux peak and the short duration until frontshield separation (in case of the assumed steep entry), the benefit of the hybrid hot structure concept is rather limited (about 5 kg with the current design). A trade-off would therefore be needed to trade the mass benefit versus the increased design complexity. It is however noted that in the case of a shallower entry the benefit of a hot structure approach will be more significant.

3. Alternative frontshield TPS material: Use of advanced ablator material currently developed by NASA in the frame of the 'Heatshield for Extreme Entry Environment Technology' (HEEET) program.

4. Backcover without hot structure: Since the backcover now is assumed to be released shortly after peak heating, the mass advantage of a hot structure concept on the

backcover is minimal. For a standard cold substructure the required ASTERM thickness has been analysed as 33mm (including 50% margin). The mass advantage of a hot structure concept would therefore only be about 1 kg, while it would still significantly increase the complexity and cost of the probe.

5. Alternative backcover TPS materials: Cork-based ablators might be a potential choice for the backcover with a better thermal performance than ASTERM. E.g. the Norcoat Liege ablator from ArianeGroup has been qualified for Exomars for heat fluxes up to about 2 MW/m^2. However, the suitability and performance under a H_2/He-atmosphere would have to be verified.

6. Increase bondline temperature limit: In case a hot structure concept is maintained, available study results indicate that the bondline temperature limit could be increased to the order 1200°C. Dedicated verification for the ablator/substructure material combination would be required.

16.1.7 Technology Needs

*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
*	Heatshield (front)	Fully-dense carbon-phenolic ablator	ArianeGroup (F)	4	3rd party	Sustainable material with large flight heritage but different environment
*	Heatshield	High-temperature bonding (>1000°C)	TECNALIA (E)	4	EU	Delta-development required for dense CP onto C-C at high temperature
*	Heatshield (back)	Low density carbon-phenolic ablator (ASTERM)	ArianeGroup (F)	4	ESA	TRL6 reached for sample return missions (Earth return)
*	RHU's	RHU's				Adaptation as needed

The aspects to be addressed on the fully-dense carbon phenolic material include in particular: 1/ verification of the capability to produce a monolithic heatshield including the spherical nose, 2/ verify sufficient micro-meteroid impact resistance, 3/ verify the material performance in plasma test under representative loads, 4/ complete the material characterisation.

16.1.8 Test Facility Needs

The response and performance of the heatshield materials will have to be verified under relevant plasma conditions, representing as closely as possible the atmospheric composition and dissociation status, the expected convective and radiative heat flux levels, as well as pressure and possibly shear loads. To the knowledge of the study team there are no plasma test facilities available capable to reproduce the relevant environment.

However, attention is drawn to the JP200 facility at ArianeGroup in Bordeaux, which is an open 20 MW Huels arc-heater facility capable to achieve in stagnation point configuration up to about 80 MW/m² at 5-50 bars (TBC). Operation would presumably be limited to air. Dedicated assessment would be required to further assess the capability and suitability of the facility. Possibly lasers could be used to additionally reproduce the radiative component of the heat flux environment.

Additional attention is drawn to the plasma facilities operated at IRS (University of Stuttgart) which are based on a magneto-plasma-dynamic generator (MPD). These facilities can be operated in both air and H_2/He. Limits in terms of heat flux conditions would have to be assessed, but might be above 10 MW/m². This may therefore allow to correlate the material performance in air versus the one in H_2/He.

16.2 Orbiter

16.2.1 Requirements and Design Drivers

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
THE-010	Temperature range for units and payloads: 0 - 30 °C	

16.2.2 Assumptions and Trade-Offs

Assumptions	
1	No sunshield is required for the RTGs (as Sun distance of S/C is always >1AU)
2	MLI effective conductivity at low T: 0.02 W/m²K
3	MLI emissivity outer layer: 0.5
4	Orbiter's outer surface (MLI area): 36 m²
5	Internal S/C temperature limits: $0 \leq T_{S/C} \leq 30$ °C
6	ΔT between payload compartment and radiator: 10 K
7	Radiator emissivity: 0.8
8	Louver efficiency: 0.7 (considered conservative)
9	"Excess power radiator" max temperature: 60 °C

16.2.3 Baseline Design

The orbiter is covered in a high-performance MLI which incorporates micro-meteoroid shielding capabilities and uses a HELPAC spacer. This MLI was developed for JUICE and has currently a TRL of 6 (for JUICE). Heat leakage through the MLI is estimated to be in the order of 120 W (0°C inside) to 140 W (30°C inside).

Three types of radiators are used to radiate excess heat. Classical (SSM-taped) radiators are sized such that in the hot case (internal heat dissipation of 465 W) the P/L compartment is maintained within a maximum temperature of 30°C.

In order to limit the demand of heater power in the cold case, the baseline design foresees to pass waste heat from the RTG's into the spacecraft. The amount of utilised RTG waste heat has been set to 159 W. Additionally electrical heaters are used to be able to control the temperature.

Additional louvered radiators are foreseen to reject the used RTG waste heat in hot conditions. The louver blades close in the cold conditions and open above a defined threshold temperature. Heritage for louvers exists from the Rosetta mission.

The third radiator, the "excess power radiator", is needed in cases where not all of the generated electrical power from the RTGs is consumed. The excess power radiator will be mounted on standoffs with its rear side covered by MLI in order to conductively and radiatively insulate it from the spacecraft. Any RTG excess electrical power will be 'burnt' in heaters mounted on the rear side of the excess radiator.

An overview of the various radiators is shown in Figure 16-10.

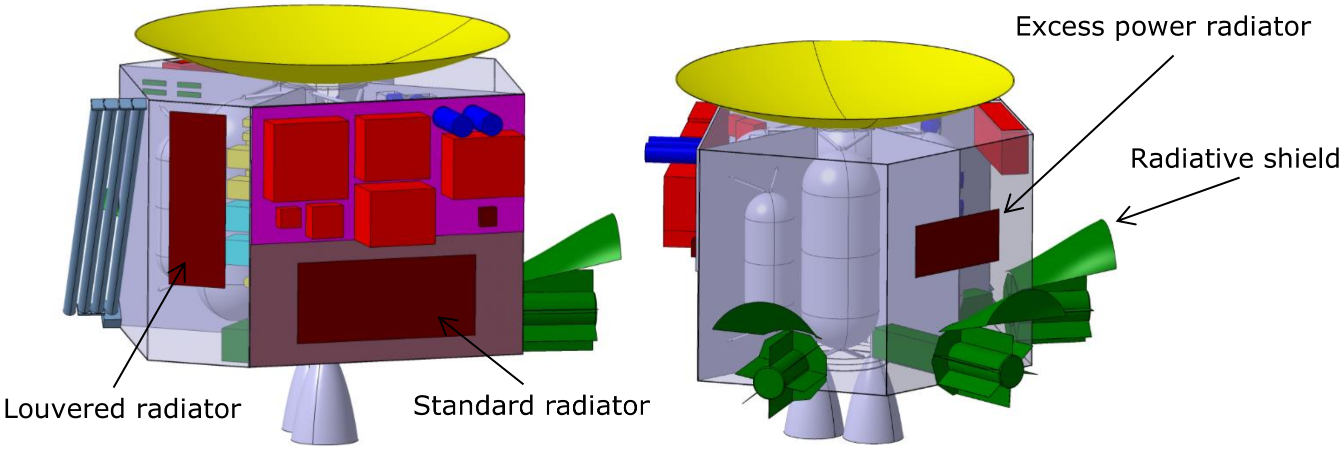

Figure 16-10: Radiators on the orbiter

The RTGs need to be thermally insulated from the S/C platform. Conductive decoupling of the RTG is achieved by designing it interface structure in Titanium (low conductivity) with long conductive paths, and by implementing a radiative shield covered by MLI.

The required electrical heater power for the cold cases depends on the internal heat dissipation. The estimated required electrical heater power for every mode can be found in Table 16-4. The electrical heater power was derived by assuming the S/C operates at 0°C in the cold case. The maximum electrical heater power needed is 48 W. At this early stage a maximum heater duty cycle of 50% is assumed. The installed heater power is therefore around 100 W.

Mode	Dissipation [W]	Heater Power [W]
CM	129	45
IMCA1	402	0
IMCA2	465	0
IN	154	20

Mode	Dissipation [W]	Heater Power [W]
IPCA	389	0
ISCom	417	0
LEOP	274	0
MM	436	0
NSM	126	48
TM	136	38

Table 16-4: Heat dissipation and electrical heater power

16.2.4 List of Equipment

The equipment needed with the respective amount and mass can be found in Table 16-5.

	Density		Amount		Mass [kg]	Margin		Mass with margin [kg]	TRL
External MLI (22-layer)	0.8	kg/m2	40	m2	32	10	%	35.2	6
Internal MLI (10-layer)	0.4	kg/m2	8	m2	3.2	10	%	3.52	
HGA MLI (10-layer)	0.4	kg/m2	4	m2	1.6	10	%	1.76	6
Normal radiator area (SSM-tape)	0.2	kg/m2	0.97	m2	0.19	10	%	0.21	
Louvres	3	kg/m2	0.68	m2	2.0	20	%	2.46	
Heaters	0.06	kg/unit	80	units	4.8	10	%	5.28	
Thermistors (triplet)	0.06	kg/unit	40	units	2.4	10	%	2.64	
White paint (HGA)	0.2	kg/m2	4	m2	0.8	10	%	0.88	
Black paint	0.2	kg/m2	50	m2	10	10	%	11	
RTG radiative shield MLI	0.4	kg/unit	2	units	0.8	10	%	0.88	6
RTG excess power radiator	4	kg/m2	0.47	m2	1.88	10	%	2.07	
Total					**59.72**			**65.90**	

Table 16-5: Equipment and mass

16.2.5 Options

There are no other options defined.

16.2.6 Technology Needs

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
*	MLI		RUAG Austria	6		JUICE MLI
*	Louvers					Heritage from Rosetta
*	RTG	Use of RTG waste heat inside the S/C				Controlled approach to be developed
*	RTG	RTG conductive and radiative decoupling				To be reflected in development of I/F structure

17 NEPTUNE AEROTHERMODYNAMICS

17.1 Aerodynamics Shape

In the Ice Giants CDF study, the outer mould line of the entry vehicle is maintained from previous, CDF studies JEP 2005 RD[47] and PEP 2010 RD[7]. The outer mould line is similar to the Galileo probe:

- The JEP/PEP shape has a nose-to-base diameter ratio of 0.41 (0.512/1.25) whereas the Galileo probe's ratio is 0.28 (44.4/126.4) . A drawback with deviating from the Galileo ratio is that this has an impact on the heritage in wind tunnel testing and flight entry. It might have a small impact on the aerodynamic drag, but larger impact on the radiative and convective heating . A more detailed trade-off on the shape of the probe shall be performed in the future.
- The base diameter (1.35) in the JEP/PEP and cone angle (45degree), which are maintained in this study is similar to Galileo
- The NASA Ice giant study in 2017 RD[1] maintained the Galileo nose/base ratio, and takes a somewhat smaller base diameter of 1.20m.

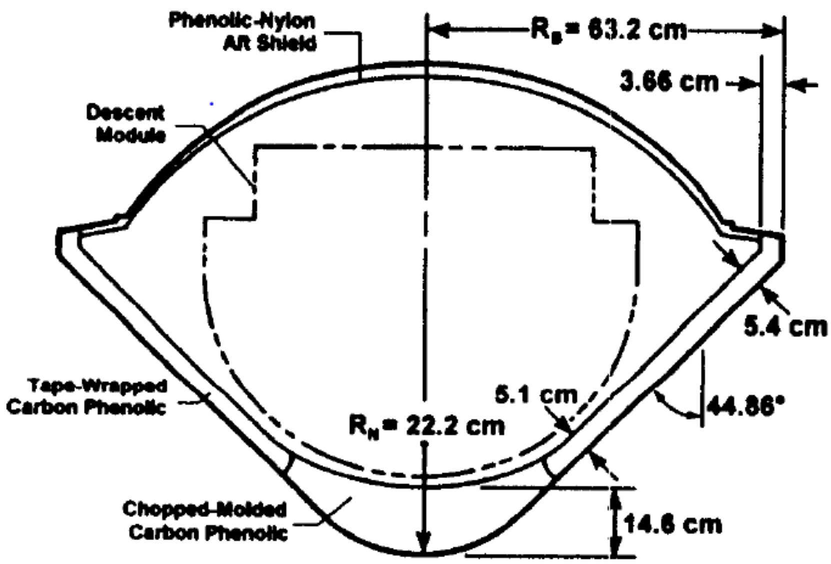

Figure 17-1: Galileo shape dimension

17.2 Aerodynamic Drag Profile

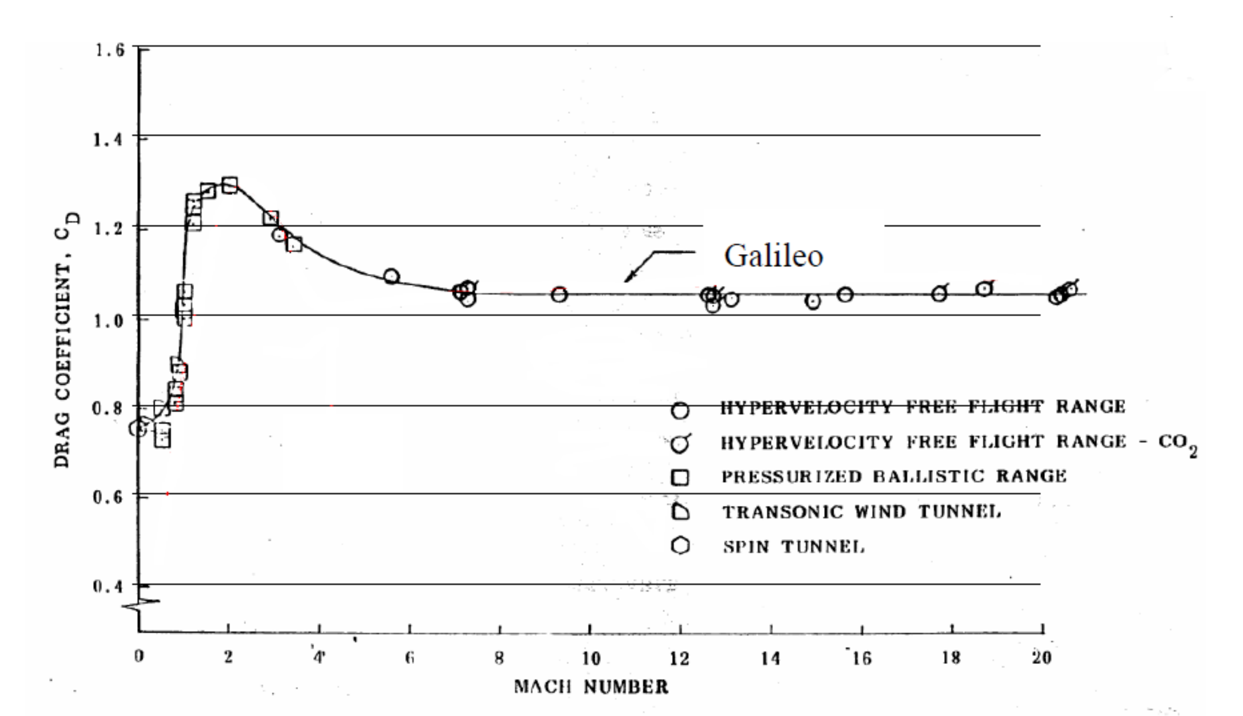

Figure 17-2: Drag coefficients of the Galileo Probe

The hypersonic drag coefficients used in this study are based on the Galileo wind tunnel experiments in the hypersonic and transonic regime. The hypersonic drag coefficient is 1.045 on which the ballistic coefficient is based. A 45 degree capsule can be made statically stable throughout the hypersonic, transonic and subsonic regime. The static and dynamic stability decreases when the sonic line of the front shield move towards the shoulder. From EVD studies RD[49], the capsule is static and dynamically stable if the centre of gravity is placed less than 26% from the nose based on the diameter. In a later stage the stability has to be studied in detail. Both the base shape and the rounding of shoulder tip are expected to influence the stability of the probe.

17.3 Atmospheric Model

The atmospheric model developed by Jean–Paul Huot RD[48] has been utilised in previous PEP and JEP CDF studies. It has no uncertainties included. In Figure 17-3 the model is compared with the more recent Neptune Gram 7 model, taken from RD[46]. The Gram model has a minimum, mean and maximum density profile indicated with resp. Fminmax=-1,0,1 . However the Gram model itself was not available at the current CDF study and in future it is needed to include this model with its uncertainties in the trajectory analysis. As it can be noticed, the range of aero-capture altitudes are below 300km where the density becomes appreciable. Lastly also note that the zero altitude is defined where the atmospheric pressure equals 1 bar.

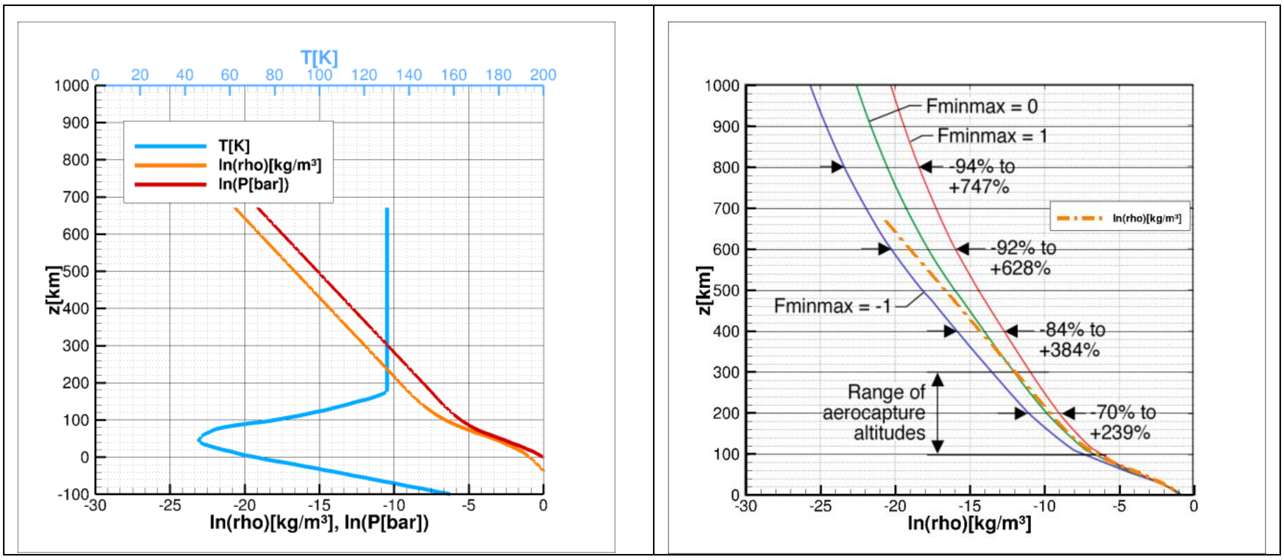

Figure 17-3: Neptune Atmospheric Model based on 85%H2/15%He (right), comparison density with the Gram Neptune 7 model (left)

17.4 Heat Flux

17.4.1 Heat Flux Correlations

The following heat flux correlations used in this study based on simulation of Simmonds and Moss. RD[50]

$$Q_{convective\ stag} = 9.08 \sqrt{\frac{1}{2\ R_N}}\ \rho^{0.419778} \left(\frac{V_\infty}{1000}\right)^{2.67892}$$

$$Q_{radiative\ stag} = 0.091\ R_N \rho^{1.3344555} \left(\frac{V_\infty}{1000}\right)^{6.75706138}$$

The back cover heat flux is considered to be 2.5% of margined convective and 1% margined radiative flux component of the stagnation point of the front shield, a percentage commonly used in pre-phase A studies for capsules, also in the PEP study in 2010 and JEP in 2005. This assumption needs to be assessed by CFD in future.

17.4.2 Heat Flux Margin Policy

The same margin policy used in PEP or JEP study has been maintained. The margin is around 1.6 times the nominal value. The number is comprised of:

- Uncertainty margin: 100%. This assumption is maintained from PEP/JEP study
- 20% reduction on both convective and radiative heating due to blockage based on Galileo studies depicted in Figure 17-4. Looking at the graph, and taking into account that the heat flux is around 100 MW/m², a radiation blockage factor of 0.4 (40 % reduction) could have been taken. However, it has been decided to keep the 20% reduction as used in PEP/JEP study
- 0 – 6% reduction on radiative component to correct the effects for the non-adiabatic shock layer compensation. Depending on the local flight conditions.

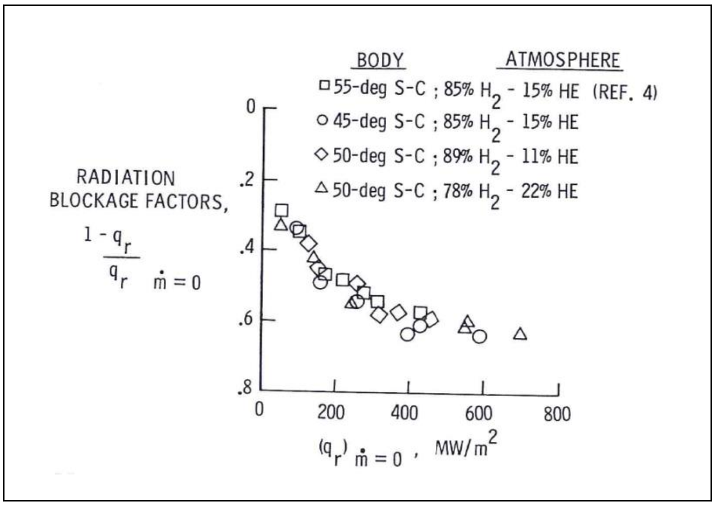

Figure 17-4: Radiation blockage factors

17.5 Entry Interface Conditions and Trajectory Analysis

The base line entry interface conditions are maintained from the PEP and JEP study. Entry state in rotating, Neptune-centered, Neptune-fixed equatorial frame. Sun and Earth directions were provided. Note that the baseline probe diameter is however enlarged with respect to the PEP case.

- Altitude [km]: 600.133
- Relative Velocity [km/s]: 23.082
- Relative FPA [deg]: -35.039
- Longitude [deg]: -8.821
- Latitude [deg]: -0.749
- Azimuth [deg]: 84.468
- Base line diameter 1.35m

Parachute opening is assumed in the following analysis at Mach =0.8 assuming a drogue and main parachute of 2 and 7m^2 area. These values are scaled up from the PEP study to meet roughly the adapted requirements. A 113 kg removal of front heat shield mass is assumed at the opening.

In Figure 17-5 the baseline trajectory is plotted, computed with the traj3d code RD[50]. It can be seen that lowering the flight path angle from 35 to 18 the maximum heat flux is reduced by half, and the radiative fraction of the total heat flux is reduced significantly at the cost of an longer flight time and 40% increased heat load.

In Table 17-1 the main characteristic of the trajectories are shown. It becomes clear that for a baseline diameter of 1.25m the maximum stagnation heat flux is 111MJ/m^2 at a pressure of 9.7 bar. No facility exists today to qualify material at these conditions since:

- Current capabilities for air can reach around heat fluxes of 50MW/m^2,
- The above facilities can not be used with H/He gas, and
- The use of H/He is expected to behave different to the commonly used gases (N2/O2/Co2).

Therefore the large uncertainty margin of 100% cannot be reduced. Even more it can be questioned whether it is sufficient, other than relying on the Galileo heritage.

A possible remedy to reduce the heat flux is to lower the flight path angle to 18 degrees. It further reduces the deceleration forces and surface pressures, which will result in a lighter structure.

On the other hand, the heat load is larger, and as a result the TPS thickness. This is not a priori major problem. The extra TPS thickness will be beneficial since it will move forward the CoG position of the capsule. This way ballast can be avoided to compensate the aft placement of the parachute assembly and payloads in the capsule.

The main drawback for lowering the flight path angle is that it makes the entry not visible from Earth.

Increasing the base diameter from 1.25 of PEP to 1.35m, and assuming the same mass, does increase the ballistic coefficient and therefore both heat flux and heat load will be slightly reduced. As a result the thickness of the TPS can also be reduced. However due the enlarged surface area increased the resulting total TPS mass might be larger in this case. Therefore reducing the base diameter to 1.2 meter is expected to reverse the effect in a beneficial way. The ballistic coefficient is based on 341 kg which is based on the TPS mass before the design freeze. Any changes to the TPS mass, increase of 18 kg, has not been flown down to the reiteration of the aerodynamic fluxes.

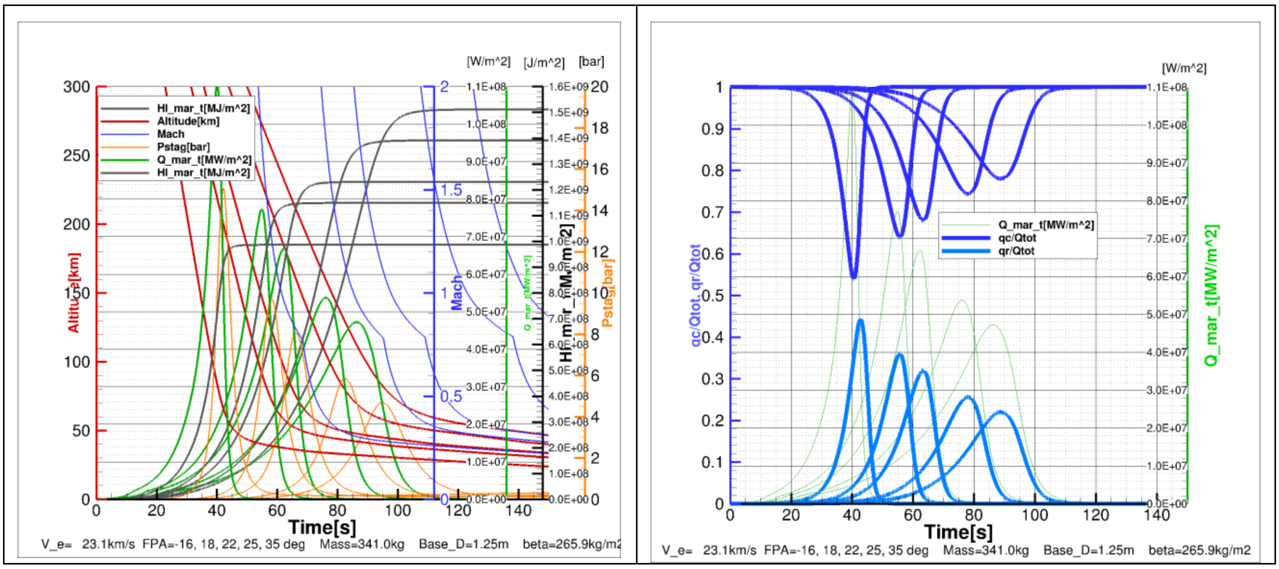

Figure 17-5: Base line trajectory plots with variation of flight path angles. On the right figure the convective and radiative ratio are plotted

base diameter	[m]	1.2	1.25	1.35	1.2	1.25	1.35
ballistic coefficient	[kg/m2]	288.57	265.95	228.01	288.57	265.95	228.01
FPA_rel	deg	-35.039	**-35.039**	-35.039	-18	**-18**	-18
Mass	[m2]	341	341	341	341	341	341
Ve	[km/s]	23.082	23.082	23.082	23.082	23.082	23.082
qmax_total	[W/m2]	121262265.3	**110908183.8**	94293856.26	57479105.45	**53735076.51**	47667001.16
qmax_c	[W/m2]	65774247.08	63032373.81	57615111.83	42935213.53	41168307.43	38013655.28
qmax_r	[W/m2]	55488018.23	47875810.02	36678744.44	14543891.92	12566769.08	9653345.874
heatload_mar_t	[J/m2]	1035987966	**987103727**	904392247	1448141328	**1391874100**	1293967362
heatload_mar_c	[J/m2]	768978466	748561517	711552037	1247027653	1212191347	1148641803
heatload_mar_r	[J/m2]	267009500	238542210	192840211	201113675	179682753	145325560
qmax_pstag	[bars]	11.157	**9.716**	8.030	3.758	**3.279**	2.611
fpa_rel_m08	[deg]	-47.39	**-47.52**	-47.52	-48.39	**-48.39**	-48.47
qmax_alt	[km]	81.85	84.40	87.45	102.66	106.15	112.14
qmax_fpa_rel	[deg]	-33.69	-33.69	-33.70	-15.19	-15.21	-15.25
qmax_time	[s]	40.2	40	39.8	76.6	76	75
qmax_vel	[km/s]	18.11	18.26	18.16	18.46	18.61	18.77
qmax_dp	[bars]	6.063	5.280	4.364	2.042	1.782	1.419
qmax_pinf	[bars]	8.18E-03	7.32E-03	6.48E-03	3.74E-03	3.33E-03	2.76E-03
qmax_mach		32.55	32.11	31.02	27.94	27.66	27.10
decmax		324.52	318.05	305.96	126.27	123.49	118.10
decmax_alt	[km]	64.65	65.50	68.53	78.09	79.75	82.38
decmax_time	[s]	42.2	42.2	42	82.6	82.4	82.2
decmax_vel	[m/s]	12.54	12.29	12.44	12.30	12.30	12.07
decmax_dp	[bar]	9.067	8.189	6.754	3.528	3.179	2.607
decmax_pstag	[bar]	16.687	15.072	12.431	6.493	5.852	4.799
decmax_pinf	[bar]	2.15E-02	2.04E-02	1.68E-02	9.76E-03	8.97E-03	7.99E-03
fpa_rel_m2		-36.04	-36.08	-36.04	-20.79	-20.91	-20.90
alt_m2	[km]	39.66	40.89	43.30	50.38	51.55	53.91
mach_m18		1.76	1.79	1.80	1.79	1.79	1.79
pdyn_m18	[bars]	0.24	0.23	0.20	0.13	0.12	0.10
pinf_m18	[bars]	0.113	0.104	0.090	0.058	0.054	0.046
pstag_m18	[bars]	0.506	0.478	0.418	0.269	0.249	0.213
rho_m18	[kg/m^3]	6.51E-02	6.02E-02	5.22E-02	3.40E-02	3.13E-02	2.66E-02
sos_m18	[m/s]	496.39	494.61	493.17	493.30	494.19	496.12
vel_m18	[m/s]	872.79	882.99	887.36	881.79	885.54	886.20
fpa_rel_m14		-37.93887215	-37.99761712	-37.95288369	-25.18408025	-25.33593791	-25.27397284
alt_m14	[km]	38.07862751	39.31802103	41.72955301	48.35257727	49.52097267	51.86380715
mach_m08		0.797375224	0.795676546	0.796492145	0.799903612	0.799282563	0.798355833
pdyn_m08	[bars]	0.067	0.062	0.053	0.043	0.039	0.034
pinf_m08	[bars]	0.152	0.140	0.119	0.095	0.088	0.077
pstag_m08	[bars]	0.23	0.21	0.18	0.14	0.13	0.12
rho_m08	[kg/m2]	8.46E-02	7.87E-02	6.84E-02	5.53E-02	5.14E-02	4.49E-02
sos_m08	[m/s]	504.21	501.77	497.79	493.75	493.02	491.97
vel_m08	[m/s]	402.04	399.25	396.48	394.95	394.06	392.76
time_m08	[s]	67.4	67.4	67.2	137	137	137.2

Table 17-1: Specific values (maximum heat flux, heat load, decelration , and potential free stream conditions at Mach 0.8 or M1.8) for the base line concept with varying base diameter at 23km/s and total weight of 341kg.

To further understand the impact of deviating from the baseline conditions, the following parameters have been varied in the trajectory analysis:

- **Total probe mass including margins** [kg]= 160.0, 180.0, 200.0, 220., 240.0, 250., 260., 278.15 288.0, 300.0, 313.34, 320.0,295.78, 330.0, 341.0, 345.0, 350.0, 360.0, 365.0, 370.0, 380.0, 390.0, 400.0
- **Relative flight path angle at entry interface:** -50, -40, -35, -30, -20, -18 , -16

- **Base diameter [m]**=1.25, 1.35. The ratio of the base-/nose- diameter is kept to 0.28 for both diameters
- **Relative Entry velocity at 600km altitude, Ve [km/s]**=21,0, 23.082, 25,7

The resulting trajectory values are plotted in Figure 17-6 to Figure 17-13 as contour lines as function of the total mass with and relative flight path angle:

- Stagnation point heat flux , including margin
- Stagnation point heat load, including margin
- Maximum deceleration druing the trajectory
- Altitude at which the maximum deceleration occurs
- At Mach=0.8, assuming the parchute will open
  - Stagnation pressure
  - Relative fligh path angle.

Figure 17-6 to Figure 17-13 have been ordered with 3 rows each increasing the entry velocities: 21 km/s, 23 and 25,7 km/s; the columns are associated to the 2 base diameters 1.25 and 1.35. In each plot 6 value markers have been placed corresponding for flight path angles (-25 and –18) and mass (341, 288 and 140 kg).

From these figures it can be concluded that

- Substantially lowering the mass below or entry velocity from the baseline, with a 35 degree flight path angle yields appreciable lowered heat load and fluxes. For example at 23 km/s, with diameter of 1.25, yield low heat fluxes (47MW/m^2) and loads (624MJ/m^2) if the mass is reduced to 140 kg. Increasing the mass to 288 kg yields 92MW/m^2 and 897MJ/m^2.
- Flight path angles of 18 degrees yield in all cases substantially lowered fluxes, at the cost of increased heat loads compared to 25 degrees. Although not visible in the plots, it has to be highlighted that this flight path angle does not allow Earth visibility during entry.
- The conditions are also given at parachute opening. Although the flight path angle at Mach 0.8 is similar, the stagnation pressure does vary up to a factor of 2 for 140 of 341 kg mass.

17.6 Potential Material Plasma Testing With H/He in Europe

It is essential to study and qualify the TPS in facilities with a hydrogen/Helium environment. The only certified facility in Europe, to the author's knowledge, able to qualify material samples for high enthalpy hydrogen-Helium mixture in Europe is the magnetoplasmadynamic PWK1 arcjet at IRS. It has been developed in the 1980s based on the know-how on electric propulsion devices. This facility was designed and certified for light weight gases, including hydrogen and could potentially test samples in the range of when using lightweight gases hydrogen for effective enthalpies in the order of 85MJ/kg with peak values 450MJ/kg. It is expected that PWK1 can reach effective enthalpies larger than 400MJ/kg, at total pressures in the hPa-range and heat fluxes 0.5MW/m^2. Further investigations to adjust performance of the facility is needed.

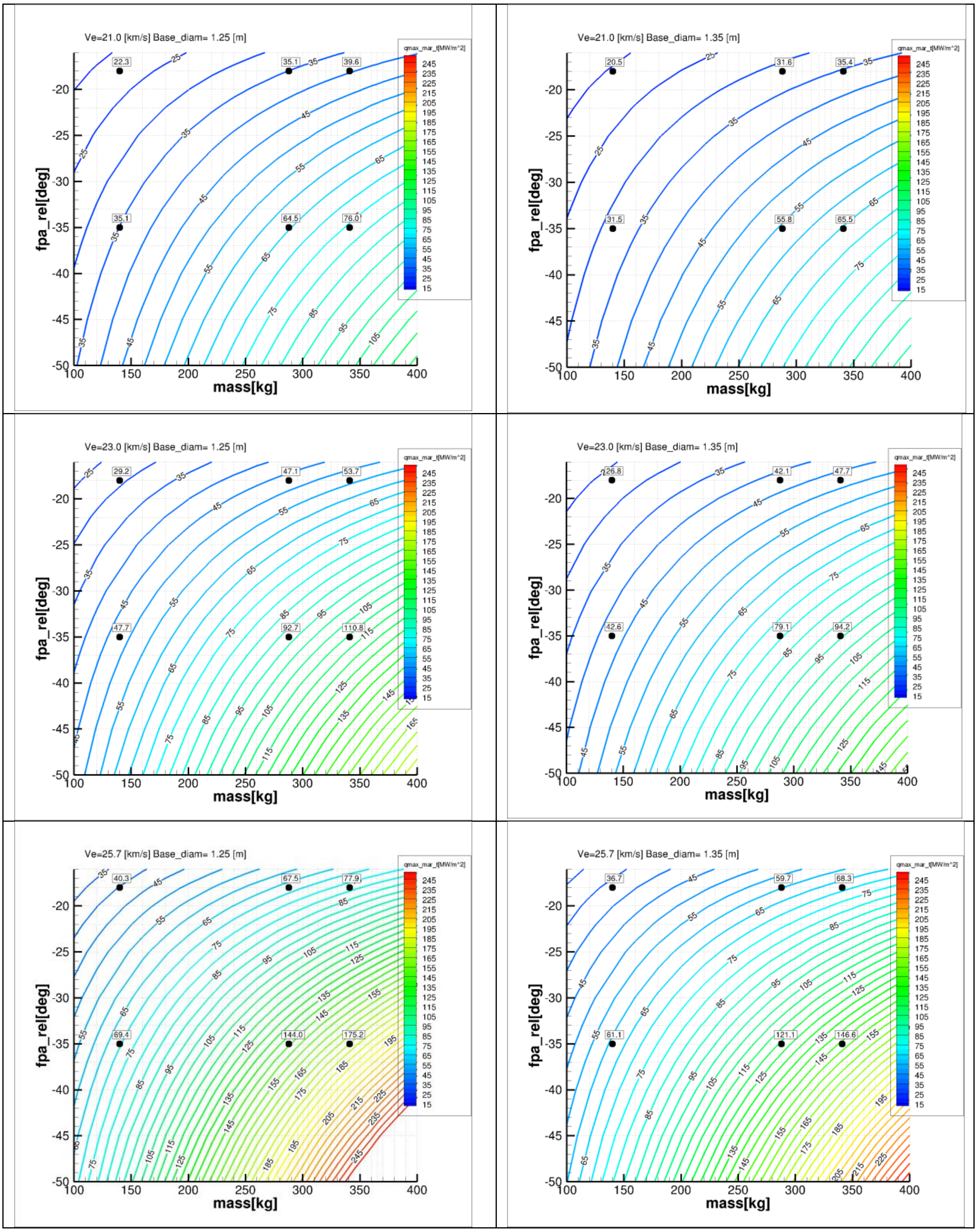

Figure 17-6: Stagnation point heat flux as function of mass and flight path angle for Ve=21,23,25 km/s and base diameter 1.25, 1.35m

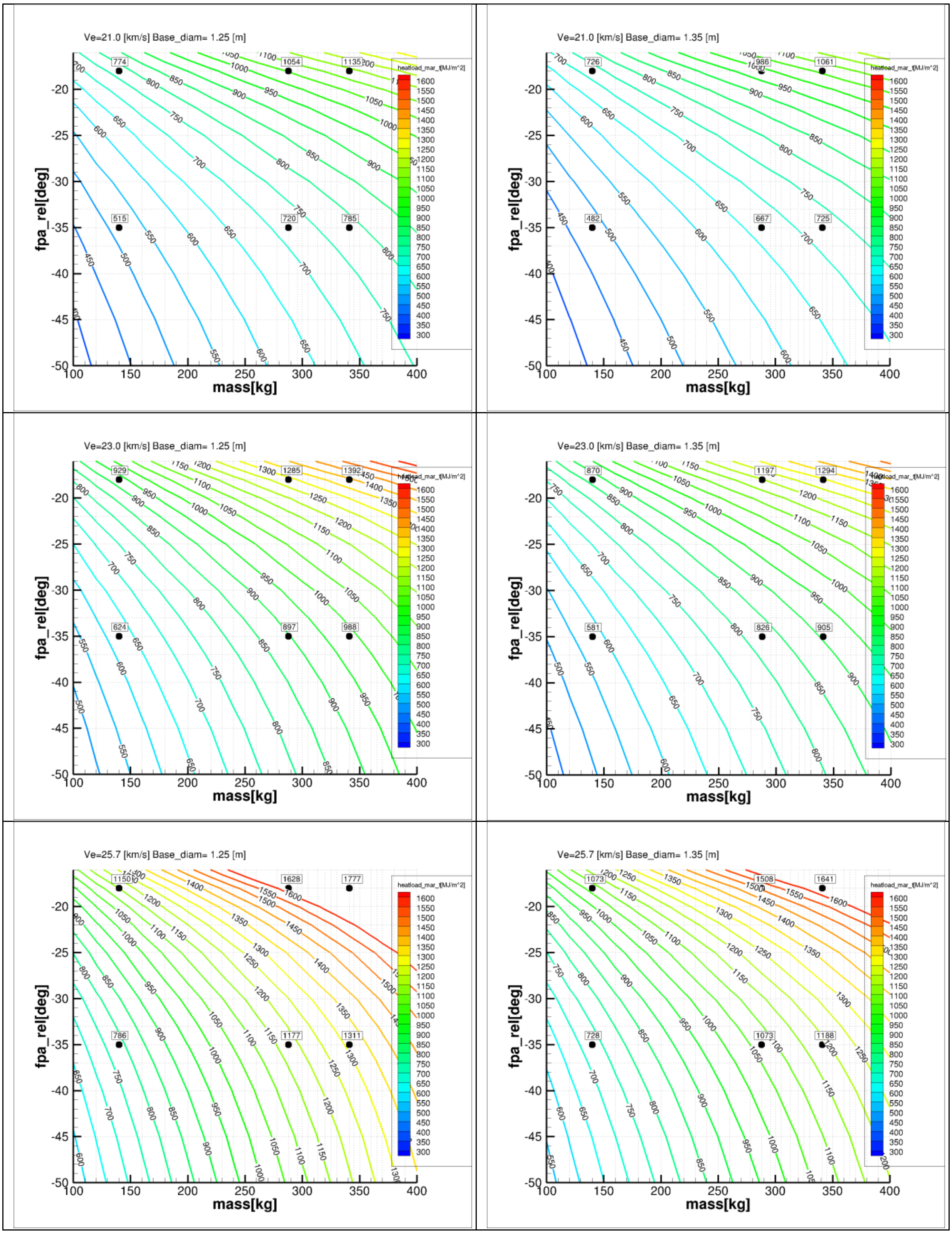

Figure 17-7: Stagnation point heat load as function of mass and flight path angle for Ve=21,23,25 km/s and base diameter 1.25, 1.35m

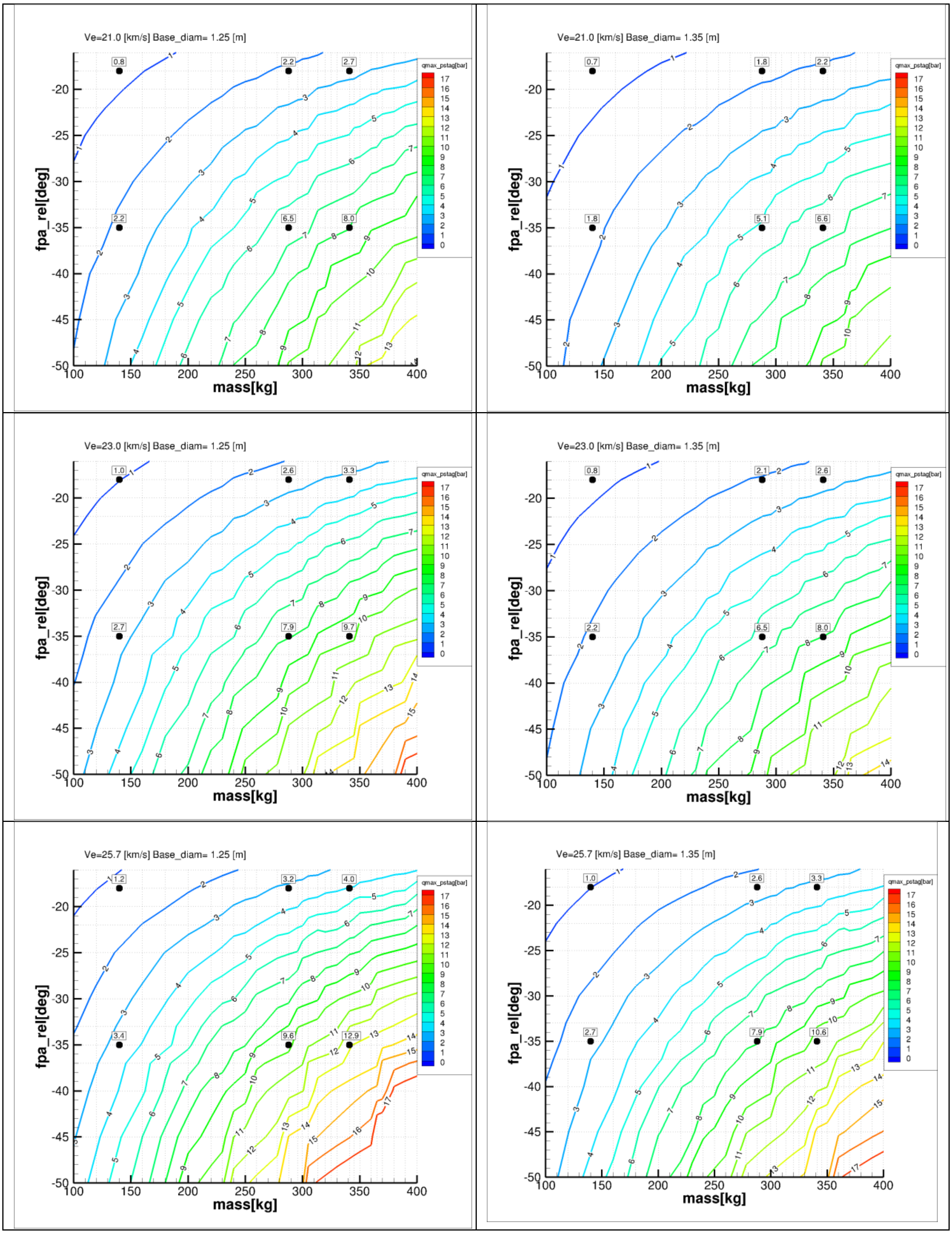

Figure 17-8 Stagnation pressure at maximum total heat flux as function of mass and flight path angle for Ve=21,23,25 km/s and base diameter 1.25 , 1.35m

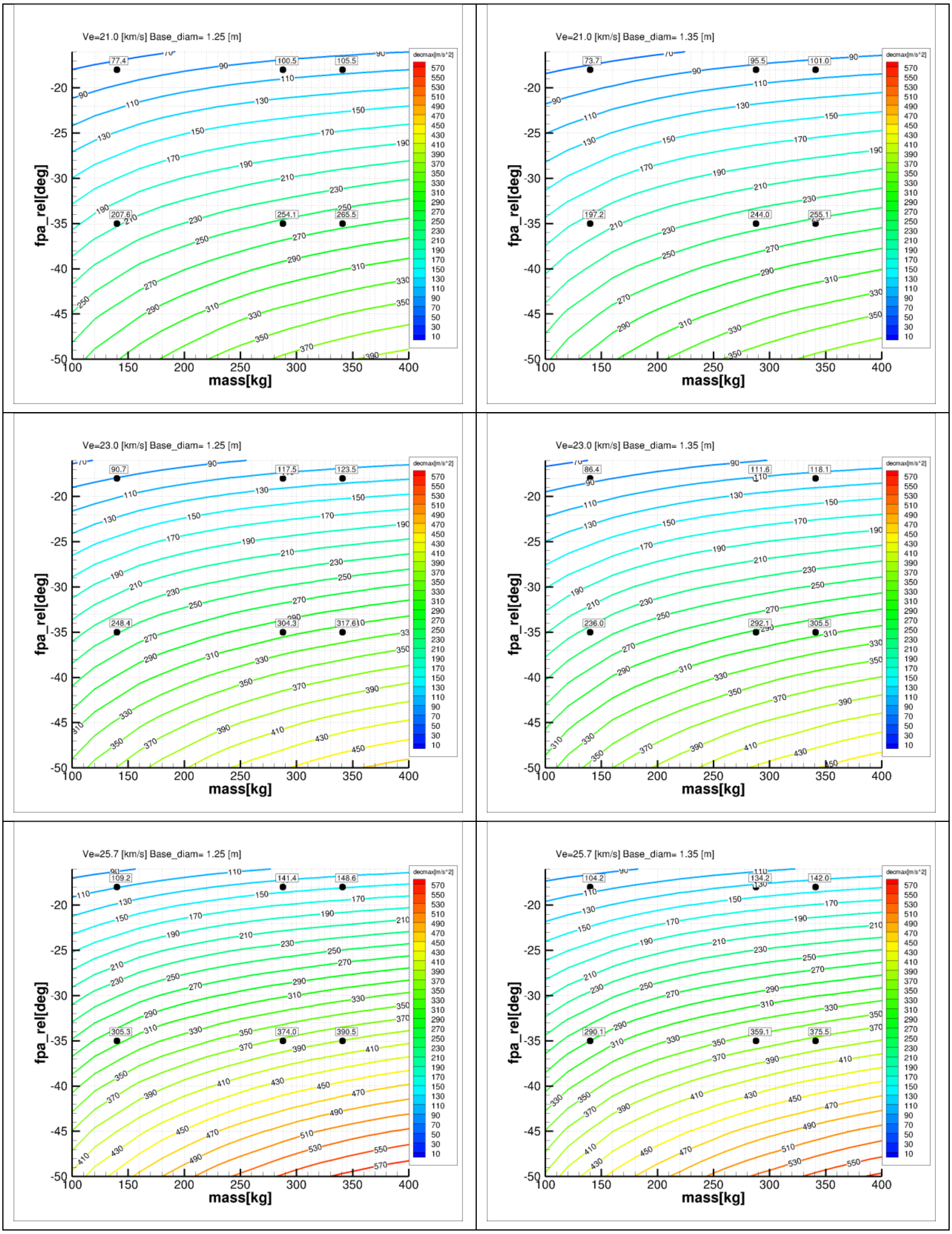

Figure 17-9: Maximum deceleration as function of mass and flight path angle for Ve=21,23,25km/s and base diameter 1.25 , 1.35m

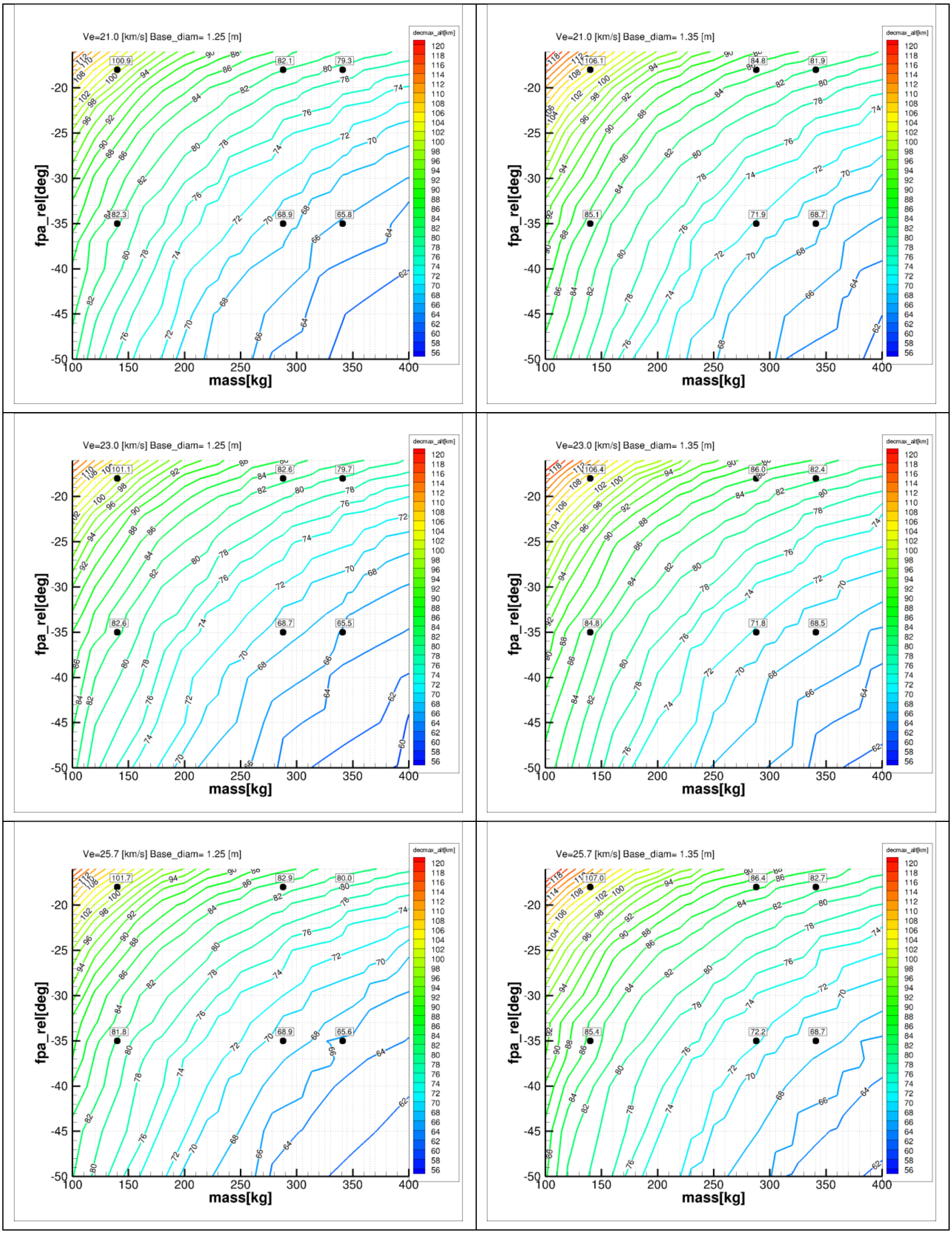

Figure 17-10: Altitude at maximum deceleration as function of mass and flight path angle for Ve=21,23,25km/s and base diameter 1.25 , 1.35m

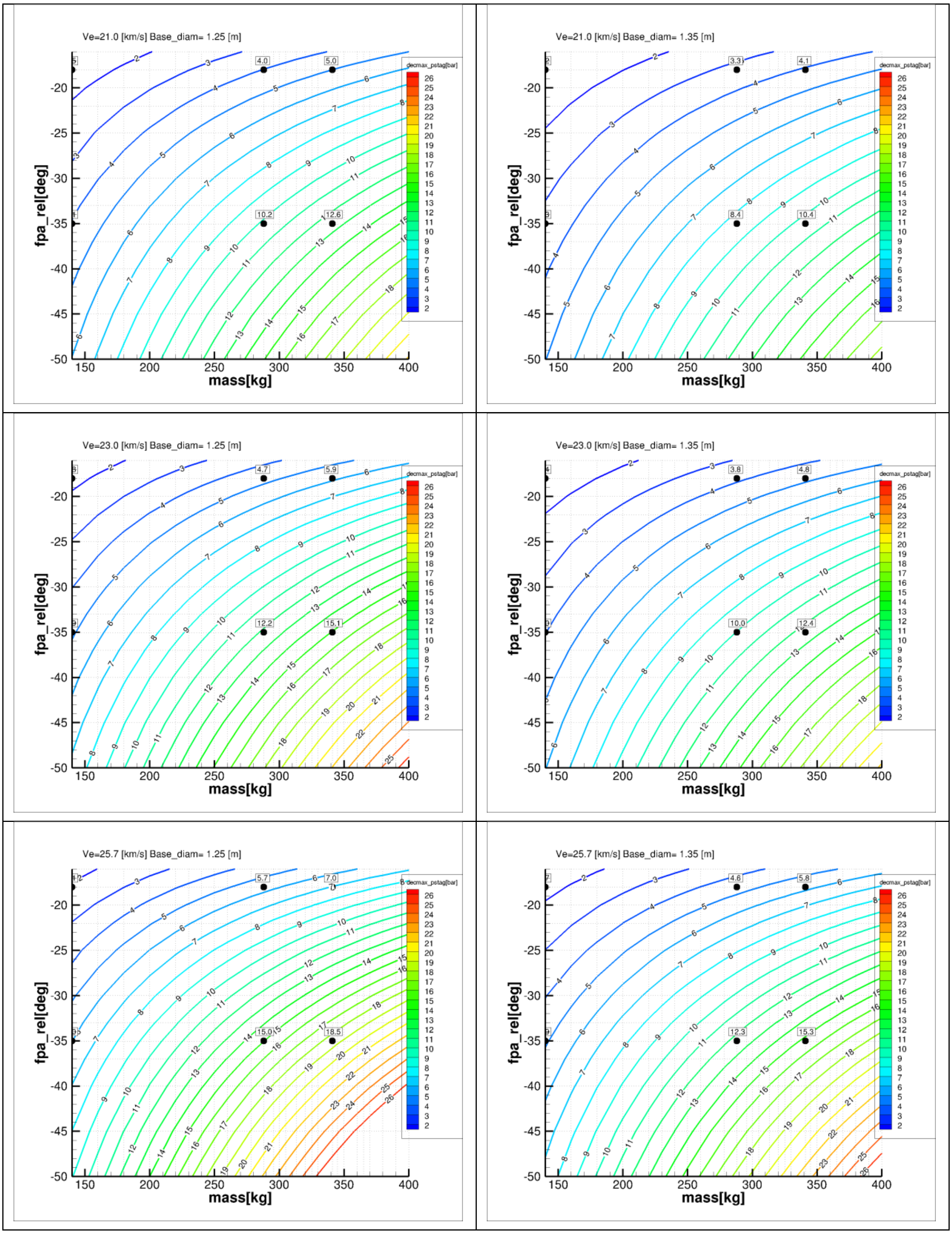

Figure 17-11: stagnation pressure at maximum deceleration as function of mass and flight path angle for Ve=21,23,25km/s and base diameter 1.25 , 1.35m

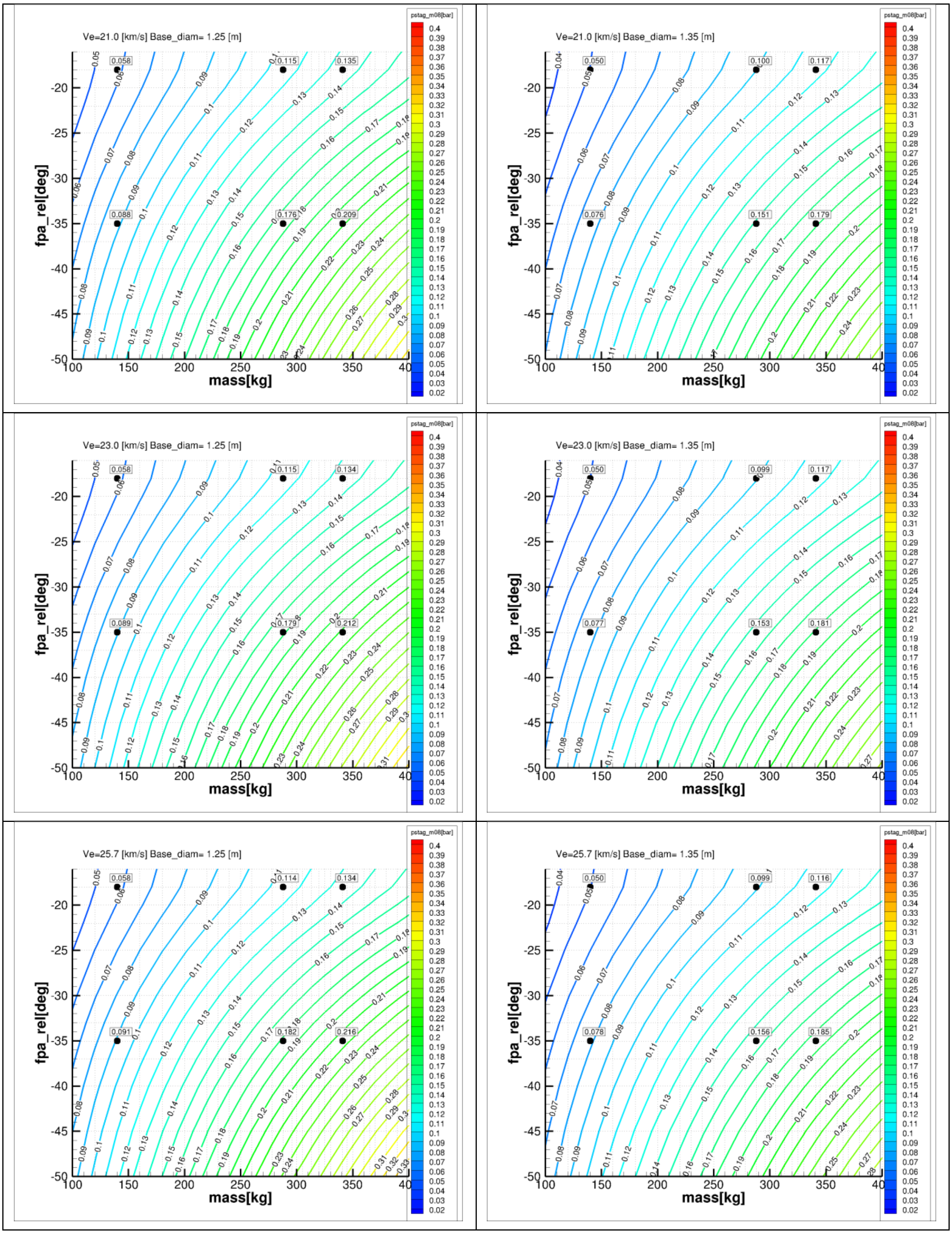

Figure 17-12: stagnation pressure at Mach 0.8 as function of mass and flight path angle for Ve=21,24,25 km/s and base diameter 1.25 , 1.35m

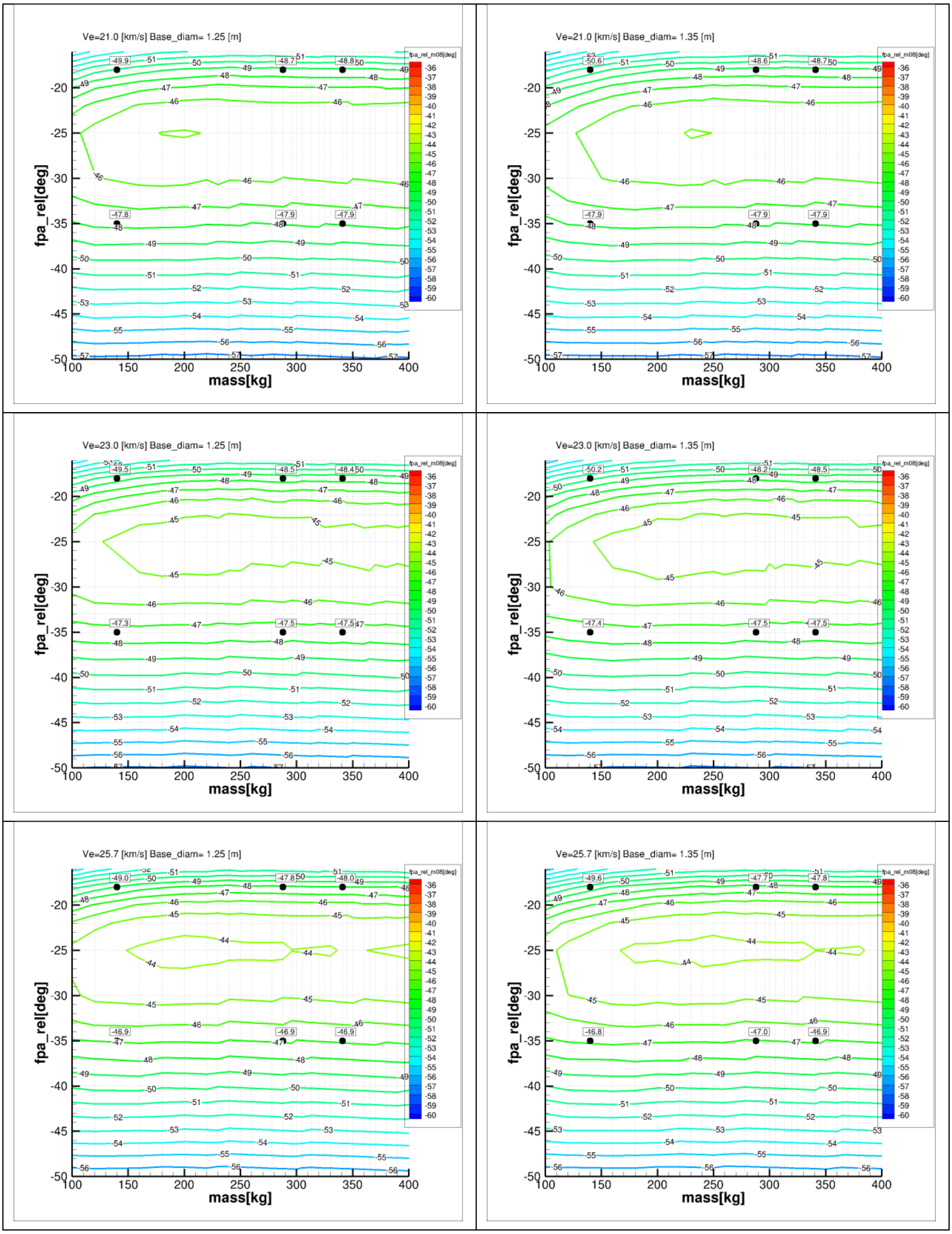

Figure 17-13 Relative flight path angle at Mach 0.8 as function of mass and flight path angle for Ve=21, 24,25 km/s and base diameter 1.25 , 1.35m

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18 NEPTUNE EDS PARACHUTE

The Entry and Descent System (EDS) described below focuses only on the Parachute Assembly System (PAS) leaving the (hypersonic/supersonic) Entry phase under the responsibility of the Aerothermodynamics (see Chapter 17).

The PAS, see Chapter 18.3, consists of a two-stage system with a mortar-ejected subsonic pilot chute that is used to deploy the main subsonic parachute.

Each parachute consists of a canopy (ribbon, suspension lines, radial tapes, vent tapes, hem tapes), a riser, 3 bridles and a confluent fitting – see Figure 18-1.

The mass (and volume) allocation for these components are therefore included in the PAS budget together with the mortar (to eject the pilot chute) and the canister (to contain the main parachute together with the mortar of the pilot).

Additional elements required for the proper functioning of the parachute system like the separation mechanisms and the cable cutter (see Chapter 18.3) are not considered part of the PAS and can be found in the Mechanisms chapter (9)

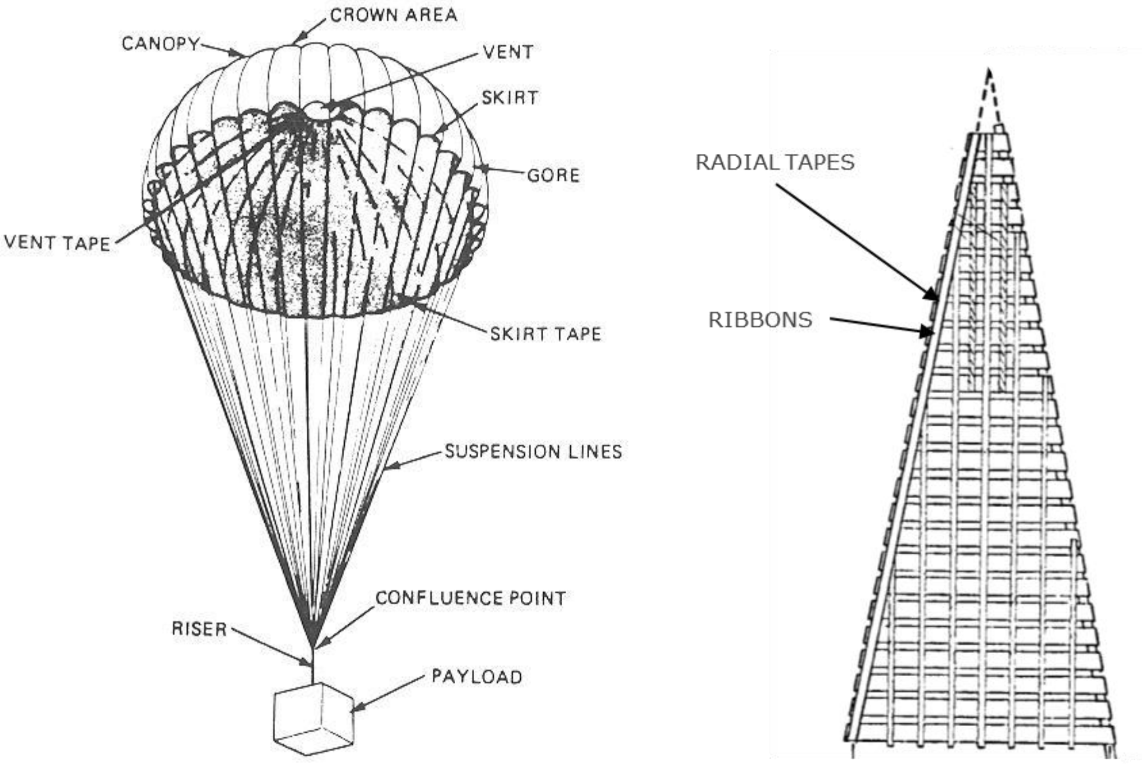

Figure 18-1: Typical parachute components (left) and a typical conical ribbon gore layout with ribbons and radial/hem/vent tapes

18.1 Requirements and Design Drivers

The main PAS requirements and design drivers are summarised below.

It is worth noting that while the release of the Thermal Protection System (TPS), especially the front shield (FS), is required to expose the scientific instruments to the atmosphere, the actual sequence and timing are free parameters that can be tuned to minimise the PAS and TPS masses and simultaneously reduce the complexity of the system.

PAS Requirements		
Req. ID	**Statement**	**Parent ID**
EDL-010	The PAS shall decelerate the entry probe after hypersonic/ supersonic phases	
EDL-020	The PAS shall allow a safe separation of the descent module and the thermal protection system consisting of the front shield (FS) and back cover (BC)	
EDL-020	The PAS shall stabilise the descent module during scientific measurement phase (1-10 bar)	
EDL-030	The PAS shall be designed to guarantee a descent from 1 to 10 bar in 90 min	SYS-020
EDL-040	The PAS shall keep the g-load to an acceptable level (<50 g)	

18.2 Assumptions and Trade-Offs

For the design of the PAS, the mass break-down reported in Table 18-1 has been assumed.

It is highlighted that the total mass of the probe at pilot chute ejection/inflation is 345 kg. After the removal of the back shell (BS) and associated structural mass, the mass left at main parachute opening is reduced to 294 kg.

Probe mass	**345**
DM mass (w/o para)	**113**
Parachute system	20
Released mass (with TPS)	213
MLI	4
TPS FS mass	127
TPS BS mass	20
TPS tot mass	**151**
Released mass (w/oTPS)	**62**
Released mass (w/o TPS) attached to FS	30.75
Released mass (w/oTPS) attached to BC	30.75
Mass released with FS	157.75
Mass released with BS	50.75
Mass at main para deploy	**294**

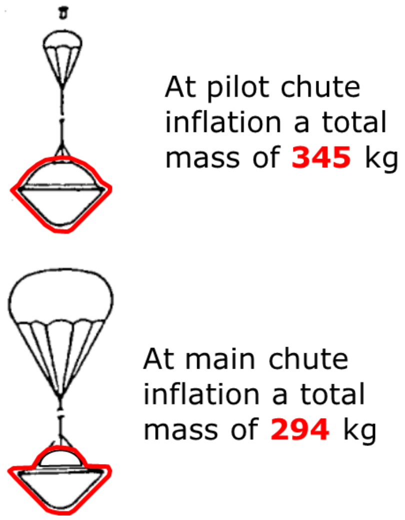

Table 18-1: Assumed mass break-down for PAS design

Moreover, assuming a stable attitude of the capsule during transonic regime, in order to limit the dynamic pressure within the values experienced in previous missions, a subsonic (Mach = 0.8) initiation has been selected.

The full list of conditions considered at pilot chute opening (see Aerothermodynamics Chapter 17) are reported in Table 18-2:

Mach	0.8
Altitude	37103 m
Atmospheric pressure	0.13 bar
FPA	-50 deg
Velocity	398 m/s
Atmospheric density	0.0731 kg/m3
Dynamic Pressure	5792 Pa

Table 18-2: Condition at pilot chute deployment

Note that, although a supersonic deployment has been avoided during the study to limit the complexity of the PAS and overall system to the minimum, present European capabilities (mostly developed within the ExoMars programme) would be available to support the introduction of a supersonic parachute.

Based on a quick comparison with a few relevant missions reported in Table 18-3, due to the high dynamic pressure at parachute opening, a conical ribbon parachute type has been selected for both (pilot and main) parachutes: the slightly lower drag of this type of parachute compared to other (e.g. disc-gap-band) is largely compensated by its performance at high dynamic pressure.

MISSION	PILOT CHUTE	MAIN CHUTE	DEPLOYMENT CONDITIONS h (km)	M	q (Pa)
Viking	None	16.2m (53 ft) Do disk-gap-band (unreefed) - mortar deployed	6.4	1.6 nominal	200 - 500
Pioneer Venus	0.76m (2.5ft) Do mortar deployed	4.94m (16.2ft) Do conical ribbon	67.1	0.8	3300
Galileo	1.14m (3.74ft) conical ribbon mortar deployed	3.8m (12.48ft) Do conical ribbon		pilot: 0.91-1.01 main: 0.87-0.97	4875 - 7648
Mars Pathfinder	None	12.7m (41.8 ft) Do disk-gap-band mortar deployed	7.5 – 12.1	1.70- 2.30	580 - 703
Cassini - Huygens	2.59m (8.5ft) Do disk-gap-band mortar deployed	8.3m (27.2ft) Do disk-gap-band	141-180	1.38 -1.73	287 - 440
MER	None	14.1m (46.3 ft) Do disk-gap-band		1.4 9- 2.30	569 - 830

Table 18-3: Previous mission and adopted parachute system (Courtesy of VOR)

The assumptions described above are summarised in Table 18-4:

Assumptions	
1	The PEP design of the probe and specifically of the parachute is kept as reference
2	The total entry mass of the probe (at pilot activation) is 345 kg
3	Considering a total mass of TPS of 151 kg, the total mass at main parachute opening is 294 kg (see detailed mass break-down in Table 18-2)
4	The parachute sequence shall be activated at Mach = 0.8 at a dynamic pressure of 5792 Pa

Table 18-4: Assumption and Trade-off

For all the design and the analysis of the two parachutes, the Parachute Engineering Tool (PET) RD[53] has been extensively used. All major outputs are reported in the following sections, while additional (minor) information is available if necessary.

18.3 Baseline Design

The PAS sequence is depicted in Figure 18-2:

- After the initial hypersonic/supersonic entry phase, the Mach = 0.8 condition is detected and the PAS sequence initiated
- At Mach = 0.8 the pilot parachute is mortar-ejected.
- With a fraction of a second delay, the back cover separation mechanisms are commanded
- The pilot chute inflates and removes the back cover that extracts the main parachute (that is attached to it)
- The main parachute is deployed and inflated
- After a short period of time required for the probe to be stabilised, the front shield separation mechanisms are commanded and the FS is detached
- The descent module under the main parachute is ready for scientific measurement acquisition.

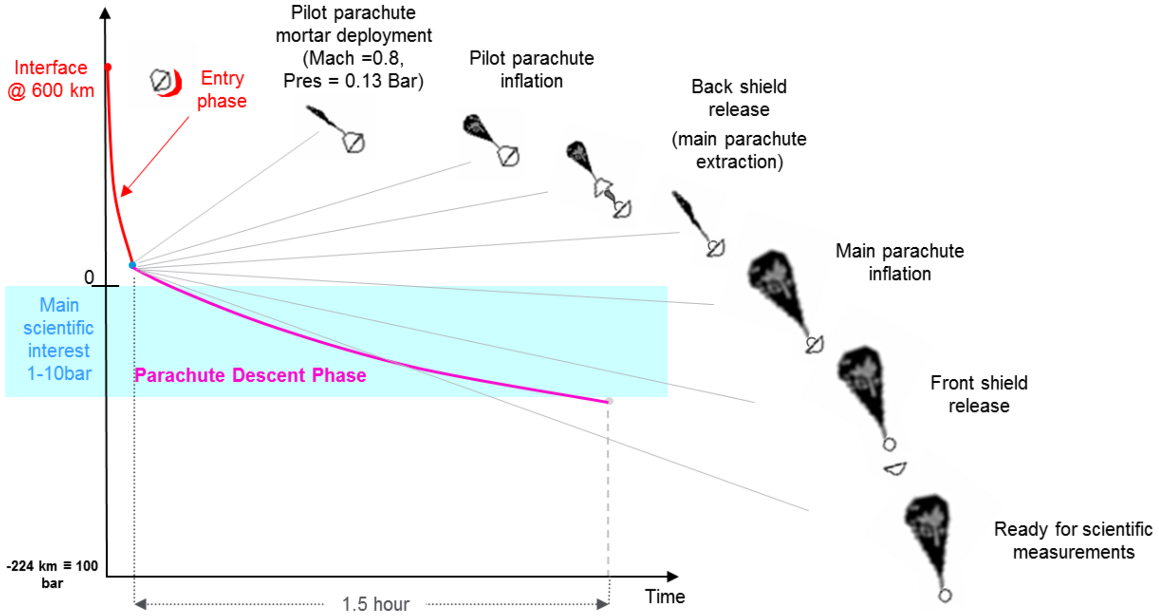

Figure 18-2: PAS sequence

18.3.1 Pilot Chute Design

The pilot chute is triggered at Mach = 0.8 (see above the detailed description of the conditions in Table 18-2) and it is needed to remove the BC and extract the main parachute.

To guaranteed sufficient drag to extract the BC, a nominal diameter Do = 1m has been selected.

With the above condition, the simulation of the parachute opening has been performed and a maximum force, experienced at parachute inflation, has been estimated to be 3620.95 N – see Figure 18-3 (left) with a maximum deceleration of 3.5 g. The velocity decreases from the initial 398 m/s to 343.5 m/s.

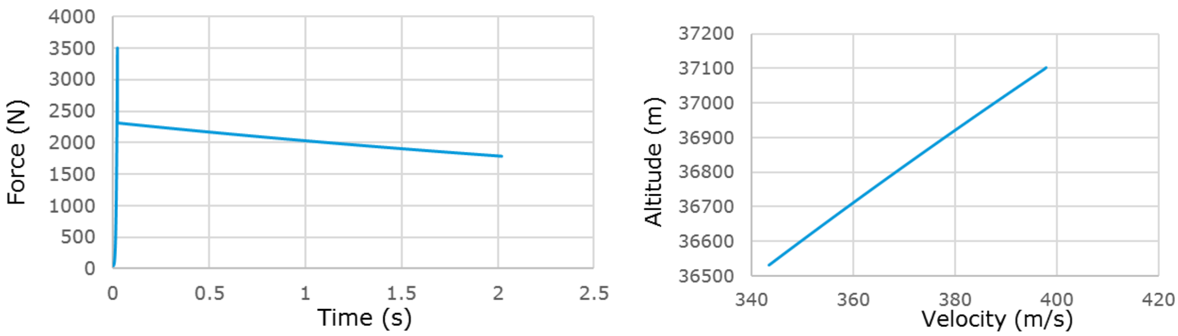

Figure 18-3: pilot chute force (left) and velocity at deployment/inflation

Additional output (for brevity not fully reported here) have also been obtained. For clarity and further reference only the states of a few parameters at the completion of the pilot chute inflation (necessary to initiate the analysis of the main parachute) are briefly reported here below:

Altitude	36530 m
Mach	0.685
Velocity	343.5 m/s
Density	0.0756 kg/m3
Dynamic Pressure	4448 Pa
Atmospheric Pressure	0.13 Bar

Table 18-5: Condition at pilot chute end of inflation

The maximum inflation load has then been used to size and select the material of the different elements of the parachute which are reported in Table 18-6.

As standard practice, the riser of 13 m (more than 10 capsule diameter) and three 1 m bridles have also been selected.

Component	Material Name	Unit Mass (kg/unit)	Quantity	Mass (kg)	Mass Margin (-)	Mass With Margin (kg)	Safety margin (-)
Ribbon	PIA-T-87130A Type XI Class 3	0.003751	12.47	0.0468	0.20	0.0561	9.42
Suspension Line	PIA-C-87129C Type XIII	0.000827	16.00	0.0132	0.20	0.0159	1.63
Radial Tape	PIA-C-87129C Type XIII	0.000827	8.00	0.0066	0.20	0.0079	1.63
Riser	PIA-T-87130A Type XI Class 15	0.013641	13.00	0.1773	0.20	0.2128	1.09
Bridle	PIA-T-87130A Type XI Class 15	0.013641	3.00	0.0409	0.20	0.0491	1.09
Vent Tape	PIA-T-87130A Type I Class 3	0.00372	0.18	0.0007	0.20	0.0008	1.09
Hem Tape	PIA-T-87130A Type II Class 1	0.00248	2.99	0.0074	0.20	0.0089	1.36
Tot				**0.2929**		**0.3515**	

Table 18-6: Pilot chute mass break-down (excluding CF, bag and mortar)

The total mass of the pilot parachute is 0.2929 kg without margin and 0.3515 kg including 20% maturity margin on all the components. Finally assuming a pack density of 600 kg/m^3 it requires a vol = 0.0005m^3

In addition, the following items are necessary to complete the pilot chute assembly:

- Confluent fitting (CF): 0.050 kg without margin, and 0.060 kg (including 20% margin)
- Bag: 0.044 kg without margin, and 0.053 kg (including 20% margin) (see Figure 18-4)
- Mortar or Parachute Deployment Device (PDD): 0.642 kg without margin, and 0.770 kg (including 20% margin) (see Figure 18-4).

Note that the mortar has been designed for a total ejected mass of 0.4 kg at of 40 m/s.

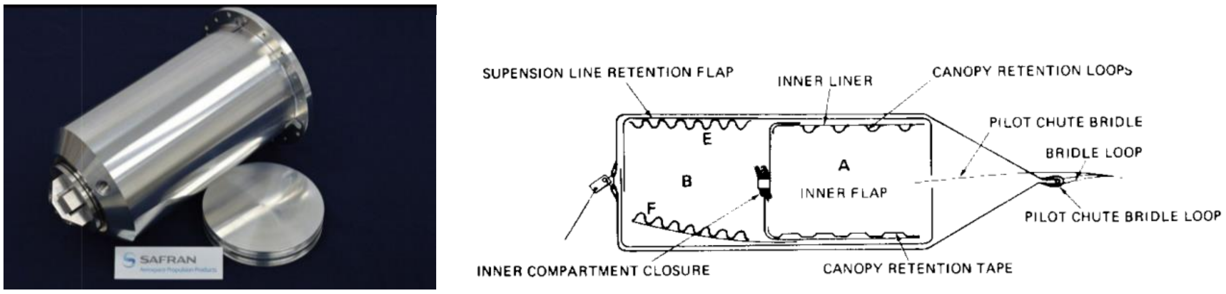

Figure 18-4: Mortar (left – ExoMars), and typical parachute bag (right)

18.3.2 Main Parachute Design

The main parachute is extracted at the time of the pilot chute inflation with the conditions reported in Table 18-5: .

To guaranteed the correct drag allowing a descent from 1 to 10 bar in 1.5 h, a nominal diameter D0 = 7.7m has been selected.

With the above conditions, the simulation of the main parachute opening has been performed and a maximum force, experienced at parachute inflation, has been estimated to be 141,464 N.

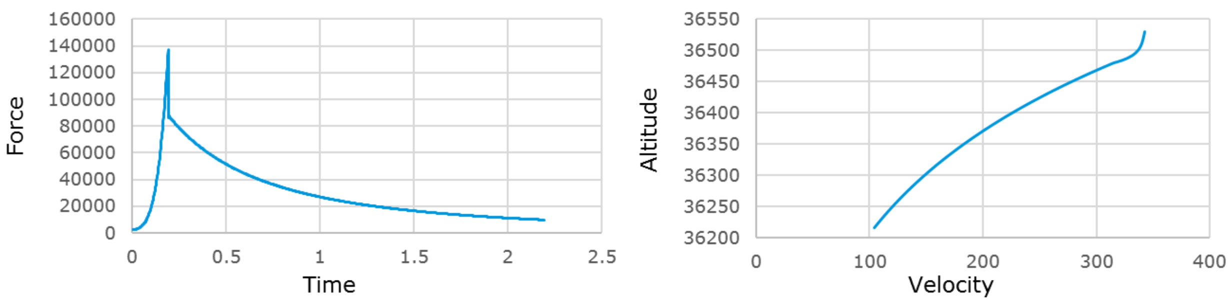

Figure 18-5: Main parachute force (left) and velocity at deployment/inflation

Note that at inflation completion, the descent module is descending at roughly 100 m/s at approximately 36000 m, guaranteeing (rough estimate) at least 6 min before reaching the target altitude for scientific data acquisition (1 bar corresponding to the target altitude, set at 0 m).

As for the pilot chute, the maximum inflation load (of 141,464 N) has been used as input to size and select the material of the different elements of the main parachute which are reported in Table 18-7.

Component	Material Name	Unit Mass (kg/unit)	Quantity	Mass (kg)	Mass Margin (-)	Mass With Margin (kg)	Safety Margin (-)
Ribbon	PIA-T-87130A Type XI Class 3	0.003751	732.61	2.7482	0.2	3.2978	2.10
Suspension Line	PIA-C-87129C Type XIV	0.014882	215.60	3.2085	0.2	3.8502	1.13
Radial Tape	PIA-T-87130A Type VI Class 6	0.011161	107.80	1.2031	0.2	1.4437	1.03

Component	Material Name	Unit Mass (kg/unit)	Quantity	Mass (kg)	Mass Margin (-)	Mass With Margin (kg)	Safety Margin (-)
Riser	6xPIA-T-87130A Type X Class 13	0.46968	7.00	3.2878	0.2	3.9453	1.26
Bridle	6xPIA-T-87130A Type X Class 13	0.46968	3.00	1.4090	0.2	1.6908	1.26
Vent Tape	6xPIA-T-87130A Type VI Class 6	0.067634	1.37	0.0926	0.2	0.1112	1.03
Hem Tape	2xPIA-T-87130A Type VI Class 8	0.034443	22.83	0.7862	0.2	0.9435	1.10
Tot				**12.7355**		**15.2826**	

Table 18-7: Main parachute mass beak-down (excluding CF, bag and canister) –

The total mass of the main parachute is 12.7355 kg without margin and 15.2826 kg including 20% maturity margin on all the components. Finally assuming a pack density of 600 kg/m^3 it requires a volume = 0.0212 m^3.

In addition, the following items are necessary to complete the main parachute assembly:

- Confluent fitting: 1.98 kg without margin and 2.377 kg - including 20% margin
- Bag: 0.632 kg without margin and 0.758 kg - including 20% margin
- Canister: 1.146 kg without margin and 1.375 kg – including 20% margin.

18.3.3 A Preliminary Timeline

With the information collected on the previous analysis, a preliminary timeline has been defined and summarised in Table 18-8 – note the close similarity with the Galileo sequence presented in Figure 18-6.

Event	Time
PC mortar deployment	$T=T_0=0$
PC inflation	$T_1=T_0+2s$
MP inflation	$T_2=T_1+2.5s = T_0+4.5s$
FS release	$T_3=T_2+10.5s = T_0+15$ s

Table 18-8: PAS timeline

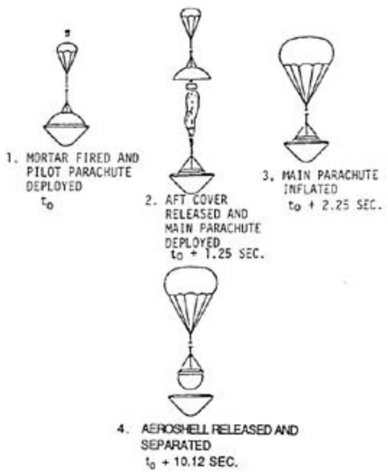

Figure 18-6: Galileo sequence

18.4 List of Equipment

The following table summarises the parachute equipment:

Pilot chute			Main parachute		
Item	**Mass w/o margin (kg)**	**Mass with margin (kg)**	**Item**	**Mass w/o margin (kg)**	**Mass with margin (kg)**
Parachute	0.293	0.351	Parachute	12.736	15.283
Bag	0.044	0.053	Bag	0.632	0.758
CF	0.050	0.060	CF	1.98	2.377
Mortar	0.642	0.770	Canister	1.146	1.375
Tot	**1.029**	**1.234**	**Tot**	**16.494**	**19.739**

Table 18-9: PAS equipment list and associated masses

The parachute system total mass is then 17.5 kg (w/o mass margins) and 21.0 kg (including 20% maturity margin for all components). Note however that, due to minor corrections noted after the design freeze of the baseline, these numbers are slightly higher than those in the baseline. As such, these numbers are higher than those in the systems chapter and were not modelled in OCDT or flown down to other subsystem's design (see Table 5-6 for the pilot chute and main parachute mass values assumed for the baseline).

The parachute system total volume has then been estimated to be 0.024 m^3 including extra 10% of needed volume due to the rather complex configuration with the mortar to be accommodated inside the toroidal bag of the main parachute (as in ExoMars mission).

18.5 Technology Needs

All the technologies identified in this study (which would be beneficial/enabler for the feasibility of this and/or other future ESA missions) adopted for the current mission are listed in Table 18-10.

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
*	Pilot Chute	PAS	IRVIN (IT/US), Pioneer (US)	9	ARD/IXV	This type of parachute was procured by IRVIN (IT/US) for ARD and Pioneer (US) for IXV. Similar capability have been developed in Arescosmos (IT) under ExoMars
*	PC Bag	PAS	IRVIN (IT/US)	9	ARD/IXV	See Pilot Chute
*	PC confluent fitting	PAS	Aeroscosmo (IT)	8	ExoMars	Modifications will be required
*	PC Mortar	PAS	APP (NL)	8	ExoMars	Modifications will be required
*	Main Parachute	PAS	IRVIN (IT/US)	9	ARD/IXV	See Pilot Chute
*	MP Bag	PAS	IRVIN (IT/US)	9	ARD/IXV	See Pilot Chute
*	MP confluent fitting	PAS	Aeroscosmo (IT)	8	ExoMars	Modifications will be required
*	MP Canister	PAS	Frentech (CZ)	8	ExoMars	Modifications will be required

Table 18-10: PAS Technology readiness

URANUS

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19 URANUS MISSION ANALYSIS

The Mission Analysis work is based on the Dual Spacecraft, Single Launch scenario from Appendix A6 in RD[1], assuming launch in February 2031 on a SLS-IB heavy lift launch vehicle. Launch sends a composite of the Uranus and the Neptune missions directly to Jupiter together with a SEP stage. The Neptune mission separates on the transfer to Jupiter.

The Uranus and Neptune missions perform independent Jupiter swingbys in December 2032. The Uranus mission performs a very close swingby at a perijove altitude of 10,000 km (see Figure 19-2), the Neptune mission a much higher one at around 857,000 km, after which the two probes travel in completely different directions, reaching their targets in April 2042 and September 2044, respectively.

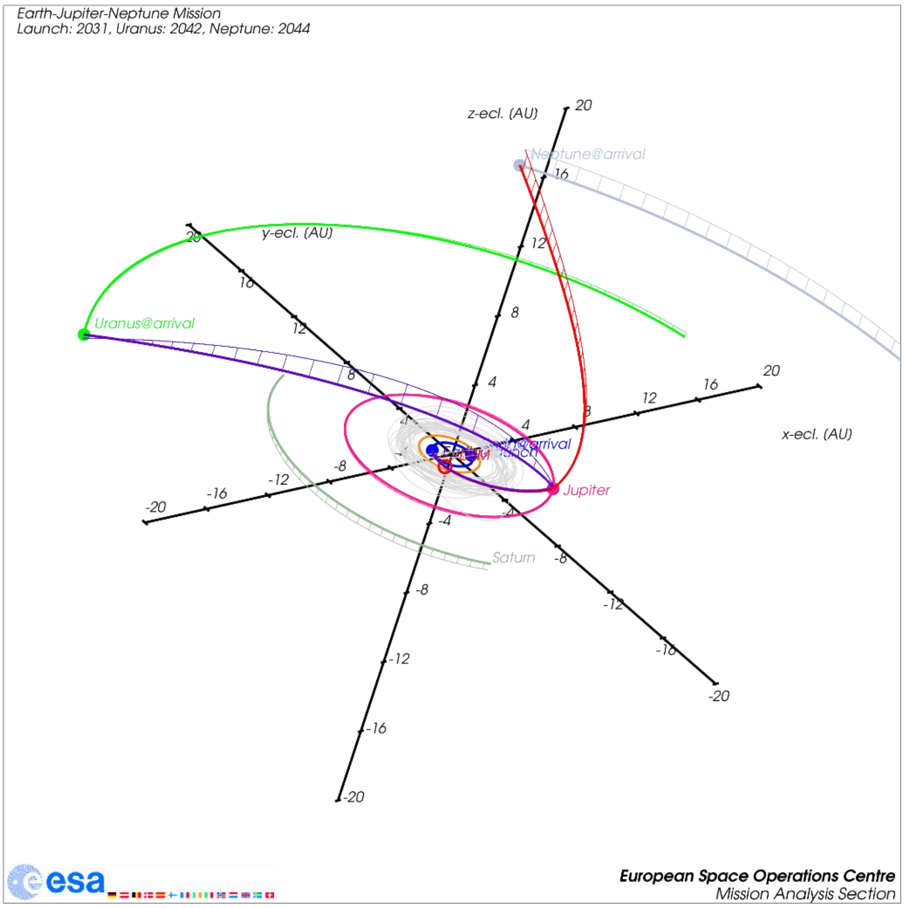

Figure 19-1: Dual Spacecraft, Single Launch Transfer Overview

The obtained arrival conditions at Uranus (considered here) and Neptune are the main input for all further analysis. The transfer scenario, timeline and arrival conditions would be significantly different for different assumptions on the overall mission.

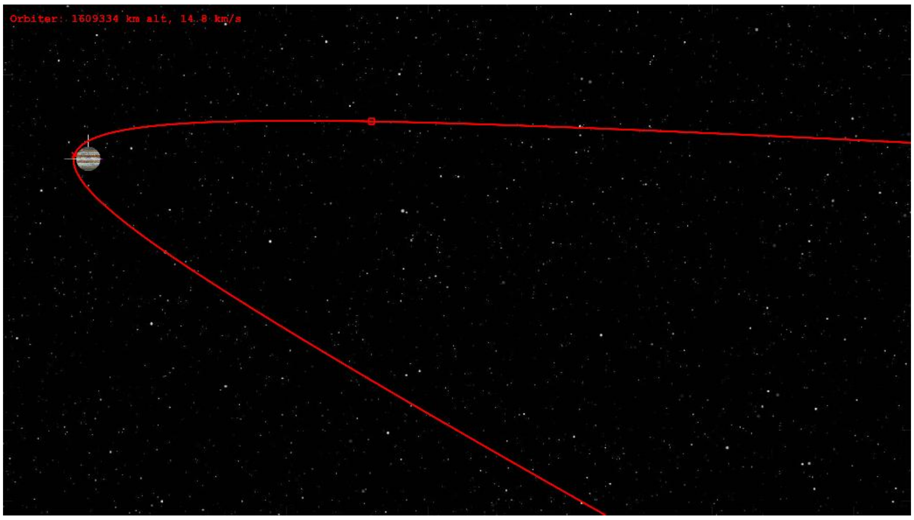

Figure 19-2: December 2032 Jupiter Swingby of the Uranus Mission

19.1 Atmospheric Probe

19.1.1 Requirements and Design Drivers

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
MA-010	Consistency with the entry conditions assumed in the earlier PEP study RD[7], specifically, a FPA relative to the rotating frame of -35 deg at an EIP altitude of 600 km above the 1 bar radius	
MA-020	The atmospheric part of the probe mission shall take place during local daylight and with visibility from the Earth.	
MA-030	The atmospheric phase of the probe mission shall last up to 90 minutes	

19.1.2 Assumptions and Trade-Offs

Assumptions	
1	If ESA provides a probe this is assumed to be carried by a NASA-provided orbiter which will target an inclination of either 73 or 107 deg with respect to the Uranus equator plane.
2	The inclination requirement of the orbiter together with the given entry FPA limits the number of possible entry locations. One possibility shall be chosen for further analysis.

19.1.3 Baseline Design

The Uranus entry diagram for the given scenario is displayed in Figure 19-3. Due to the unusual orientation of the planet's rotation axis, the Sun and Earth direction and the direction towards the incoming probe are all above the Northern hemisphere. The Sun and Earth visibility terminators are shown. All entry points above the Earth visibility terminator have Earth visibility at entry. The further an entry location is from the Earth visibility terminator, the longer its atmospheric mission can be before geometrical visibility from the Earth is lost. The minimum required Earth aspect angle must be studied in detail, using Figure 19-5.

Entry locations and directions of flight are shown for inertial flight path angles of -25, -35 and -45 deg. For -25 deg, all entry points are either beyond the Sun and Earth visibility terminator or close to it, so a daylight mission with Earth visibility would be impossible. An inertial entry flight path angle of -35 deg appears to allow missions that are consistent with the Sun and Earth visibility requirement. In the present study a relative entry flight path angle of -35 deg is required, which translates into some offset in the inertial FPA. However, Figure 19-3 is still qualitatively applicable.

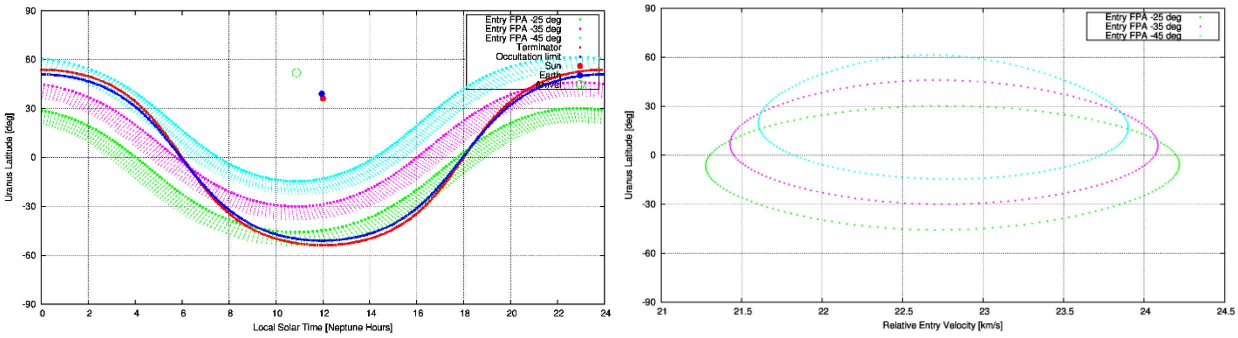

Figure 19-3: Entry Plot and EIP Velocities for 2042/4/6 Arrival at Uranus

19.1.4 Budgets

Table 19-1 lists the entry conditions for each of the four possibilities, two prograde and two retrograde. 73/1 and 107/2 are daylight missions, and out of these two, 107/2 has the lower entry velocity, so this option is retained as reference for the aerothermodynamics analysis.

Case	73/1	73/2	107/1	107/2
Altitude [km]	600			
Date/Time	2042/4/6 12:00:00 TDB			
Longitude [deg E]	24.5	135.5	-146.6	-39.4
Latitude [deg [N]	-23.7	42.0	38.4	-26.0
Velocity [km/s]	23.364	23.374	22.068	22.071
Azimuth [deg]	155.1	151.2	196.6	193.4
Rel. FPA [deg]	-35			

Table 19-1: Possible Entry Points for 73 and 107 deg inclination and rel. FPA -35 deg

Note that the longitude value given here applies only to entry at the stated epoch. The entry longitude can be modified at negligible delta-v cost just by changing the arrival time by +/- 9 hours, which will not affect any of the other parameters. Conversely, the entry latitude can be changed only by applying a steeper or shallower entry FPA or by a significant change in the arrival date, all of which would have a significant effect throughout the mission design.

Even if there is Earth visibility, an array of terrestrial radio telescopes will at best only be able to capture the carrier signal. Data transmission will have to take place via the orbiter, which will be performing UOI while the probe is performing atmospheric entry. The determining parameter for the coverage quality is the periapsis altitude of the orbiter.

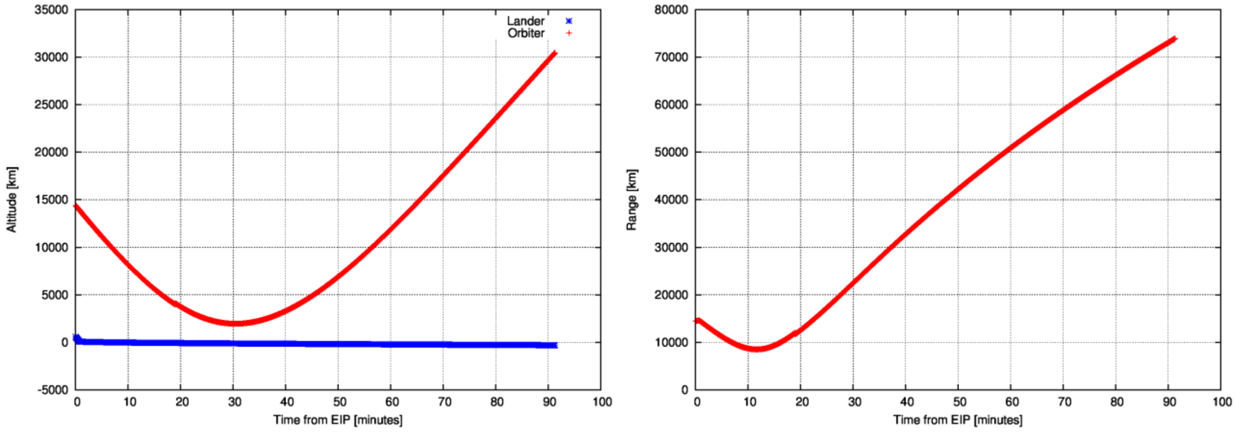

Figure 19-4: Altitudes and Slant Range, Target Periuranion Alt. 2000 km

For a targeted periuranion altitude of 2000 km above the 1 bar radius the altitude of probe and orbiter and the probe-orbiter slant range are shown in Figure 19-4, while Figure 19-5 gives the evolution of Earth aspect angle (EAA) and Earth aspect angle (OAA), i.e., the angle between the symmetry axis of the entry probe and the directions to Earth or orbiter. The probe symmetry axis is assumed to be aligned in the opposite direction of the current relative velocity wrt. the rotating Uranus atmosphere.

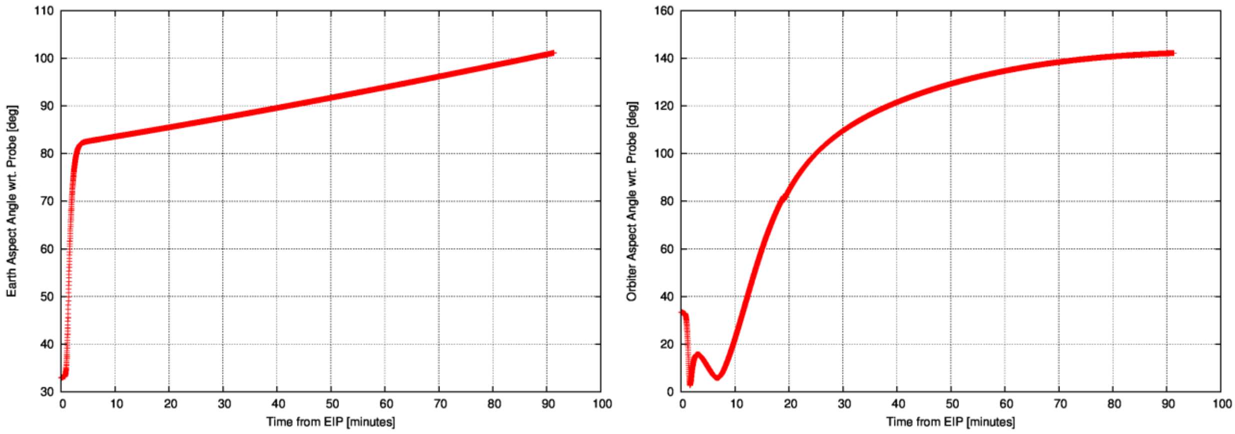

Figure 19-5: EAA and OAA, Target Periuranion Altitude 2000 km

Following entry, the probe will slow down through aerodynamic drag and its relative flight path angle will quickly deepen from -35 deg to near-vertical. As a consequence, the EAA undergoes a strong initial increase, followed by a slow drift caused by the probe being carried along by the rotating planet.

Conversely, the OAA initially approaches zero as the orbiter, which was trailing the probe on a higher and slower orbit, catches up and passes the probe directly above. Around 10 minutes after entry, the OAA increases fast as the orbiter races ahead while the probe is moving only slowly with respect to the atmosphere and its lateral motion is due only to the rotation of the latent. The OAA goes above 90 deg around 22 minutes after entry. Depending on the opening angle of the probe antenna pattern the orbiter will lose contact at the latest at that point, or likely some time earlier. A target orbiter periuranion altitude of 2000 km is inconsistent with a probe mission duration of 90 minutes

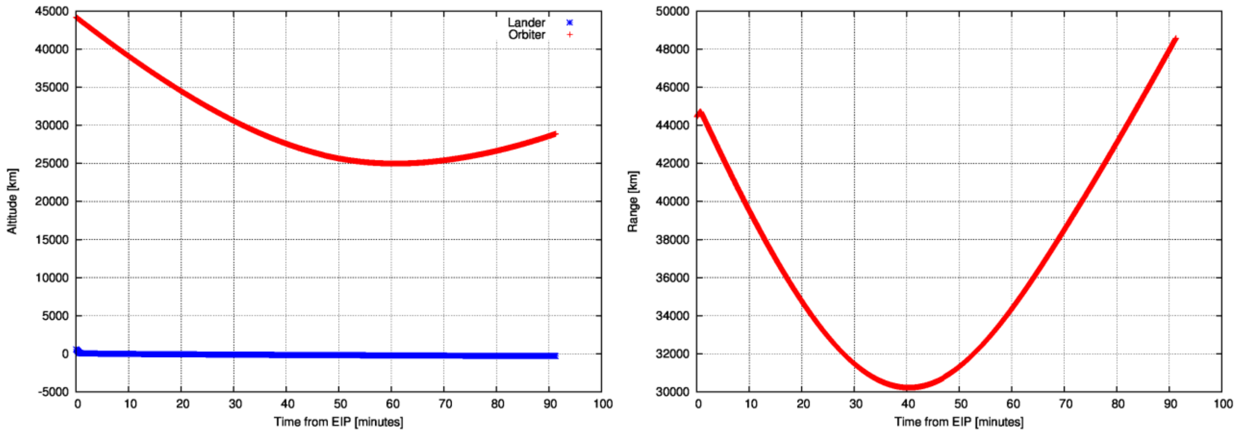

Figure 19-6: Altitudes and Slant Range, Target Periuranion Alt. 25000 km

The same set of diagrams has been produced assuming a target periuranion altitude of 25000 km. In this case, the orbiter arrival is delayed significantly and the probe-orbiter geometry is much different, leading to a much larger slant range with a minimum of 30000 km but also a time of 90 minutes from entry to the point where the OAA reaches 90 deg. This indicates that the target periuranion altitude must be at least 25000 km to support a 90 minute probe mission, possibly higher.

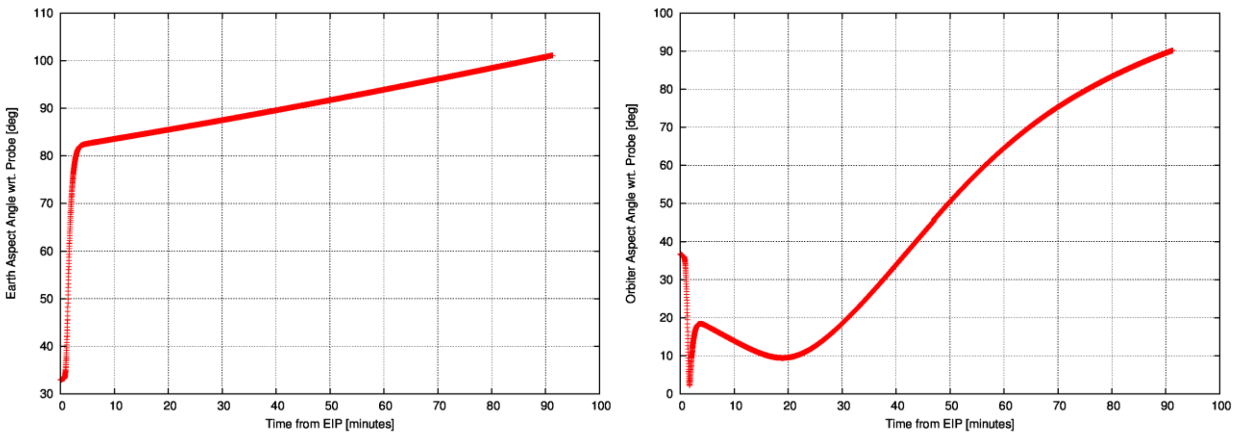

Figure 19-7: EAA and OAA, Target Periuranion Altitude 25000 km

Note that the switch to a high periuranion altitude has implications not only on the ODM and UOI size but that it may also lead to intersection of the rings. This must be studied in detail.

19.2 Orbiter

19.2.1 Requirements and Design Drivers

19.2.2 Assumptions and Trade-Offs

	Assumptions
1	In the present study, a target periuranion altitude of 2000 km above the 1 bar radius and an apouranion radius of 225 RU has been assumed. This would lead to an orbital period of 148 days, which exceeds he requirement of 140 days, though the difference in delta-v is small.
2	For the ESA-provided orbiter, the communications with the entry probe, which will then not be provided by ESA, is not assumed to be object of the study
3	The tour design is assumed not to be object of the study. The information related to the tour contained in the NASA document RD[1] is considered to be applicable.

19.2.3 Baseline Design

For the orbiter study, no considerations related to deploying a Uranus entry probe have been taken into account. Therefore, the ODM is not budgeted. The UOI manoeuvre has been modelled for different T/M ratios, assuming a target apouranion radius of 225 RU.

19.2.4 Budgets

Table 4-2 shows the UOI size and duration obtained via numerical propagation of the trajectory as function of the *thrust/mass ratio at start of the manoeuvre*, assuming that the thrust acceleration is anti-tangential to the current uranocentric velocity. The target periuranion altitude is assumed as 2000 km.

This flyby altitude would be achieved if no manoeuvre took place; it would also be the periuranion altitude of the capture orbit if the manoeuvre were near-impulsive. The longer the manoeuvre duration, the more the osculating periuranion is lowered during the burn. This, together with the significant gravity losses, should be taken into account when designing the propulsion system.

As discussed, the assumed target aopuranion radius is 225 RU. The change in the UOI when targeting to a slightly higher or lower apouranion is negligible. All further details on the tour timeline and manoeuvre sequence are beyond the scope of the CDF study and should be taken from RD[1].

Description:	**Target periuranion altitude 2000 km, apouranion radius 225 RU**	
Thrust/Mass ratio [N/kg]	**UOI [m/s]**	**Duration [s]**
0.25	1878	5658
0.5	1744	2678
1.0	1699	1373
1.5	1690	872

Table 19-2: UOI Size as Function of T/M Ratio

20 URANUS SYSTEMS

20.1 Atmospheric Probe

20.1.1 Mission & System Requirements and Design Drivers

The requirements and design drivers for the Uranus probe design case are the same as for the Neptune case, apart from the replacement of SYS-010 in Table 4-2 by the requirement in Table 20-1.

System Requirements		
Req. ID	**Statement**	**Parent ID**
SYS-010	The probe shall be carried by the NASA orbiter to Uranus	

Table 20-1: System requirements

20.1.2 Mission System Architecture

The system architecture and design of the probe is the same as for the Neptune option.

Note that, given the spin properties of Uranus, the entry and descent profiles look very different to those reported for Neptune (see Mission Analysis chapter). Nonetheless, as shown in Figure 20-1, the selected FPA of -35deg still satisfies the requirements for almost all cases.

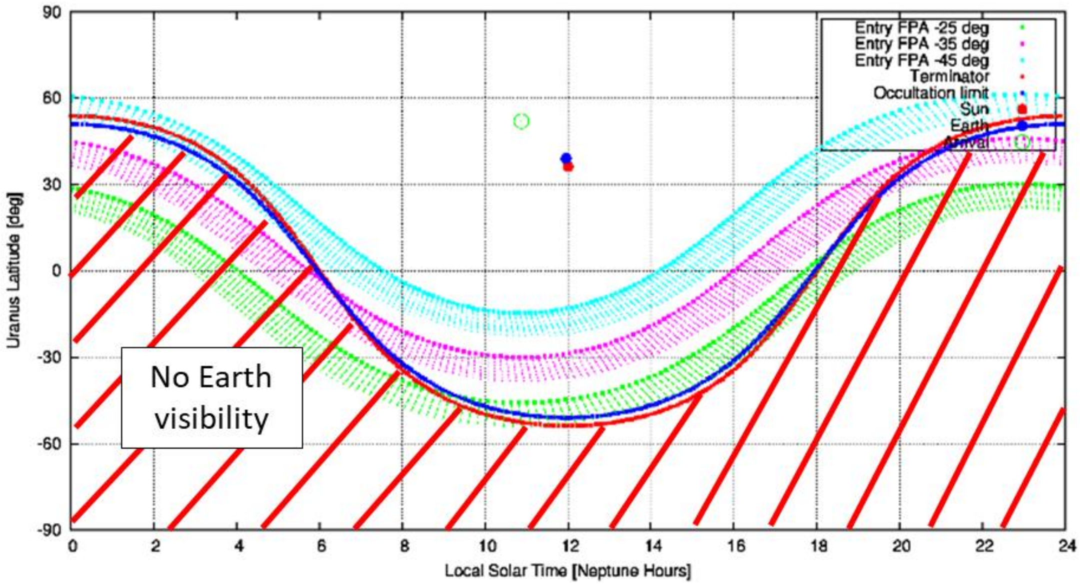

Figure 20-1: Entry conditions for Uranus probe for different FPA

20.1.3 System Baseline Design

Mass, power and data budgets for the Uranus probe are the same as for the Neptune probe.

20.2 Orbiter

20.2.1 System Requirements and Design Drivers

Mission Requirements		
Req. ID	**Statement**	Parent ID
MIS-010	The mission shall be launched within a timeframe of 2029 to 2034.	
MIS-020	The mission shall be compatible with launch via a SLS Block 1B.	
MIS-030	The total mission cost shall be within an M-class ESA science mission budget.	
MIS-040	The mission shall be capable of performing in-situ and observational measurements at Uranus with a payload suite defined by the Study Science Team.	
MIS-050	The mission shall be capable of performing in-situ and observational measurements at the Uranian moons with a payload suite defined by the Study Science Team.	
MIS-060	The mission shall include at least 2 years (TBC) of science operations at Uranus.	

System Requirements		
Req. ID	**Statement**	**Parent ID**
SYS-010	The orbiter shall be compatible with a dual launch with a NASA orbiter on an SLS Block 1B in a TBD configuration.	
SYS-020	The orbiter shall be delivered to NASA for final integration onto the launcher.	
SYS-030	The orbiter delivery date to NASA shall be compatible with the selected launch date and any pre-launch activities agreed between ESA and NASA.	
SYS-040	The orbiter shall be compatible with a storage of TBD months before integration onto the launcher.	
SYS-050	The orbiter design shall allow late access for integration of the RTGs under the launcher fairing.	
SYS-060	The orbiter shall include a payload suite of 116 kg (TBC).	
SYS-070	The operational lifetime of the orbiter shall be at least 13.5 years after launch (baseline: 15.5 years)	
SYS-080	The science operations phase of the mission shall be at least 2 years (baseline: 4 years)	
SYS-090	The orbiter shall rely solely on its own power source(s) during cruise with the NASA orbiter.	
SYS-100	The orbiter shall provide a SpaceWire interface to the NASA	

System Requirements		
Req. ID	**Statement**	**Parent ID**
	orbiter.	
SYS-110	The orbiter shall be asleep during cruise with the NASA orbiter, apart from periodic checkouts.	
SYS-120	The orbiter shall not require any active thermal control from the NASA orbiter.	
SYS-130	The orbiter shall separate from the NASA orbiter before the Jupiter swing-by.	
SYS-140	The orbiter shall be able to perform an independent interplanetary transfer from separation until Uranus.	
SYS-150	The orbiter shall be able to insert into orbit around Uranus.	
SYS-160	The orbiter shall be able to download all gathered science data within the nominal mission duration (TBC).	
SYS-170	The orbiter shall be compatible with all environments from integration until EOM.	
SYS-180	The orbiter shall include redundancy for all mission-critical functionalities. (TBC)	

Table 20-2: System requirements

20.2.2 Design Drivers

As for the Neptune design case.

20.2.3 System Assumptions and Trade-Offs

Assumptions as for the Neptune case, plus the additional assumption in Table 20-3.

Assumptions	
1	The Uranus case shall envisage maximum reuse of the Neptune orbiter design. As such, some systems (e.g. communications) may be oversized.

Table 20-3: Additional assumption for Uranus design case

20.2.3.1 Dual launch configuration trade-off

As for the Neptune design case.

20.2.3.2 Radiation shielding trade-off

The radiation levels observed by the spacecraft equipment behind 2.5, 4 or 10 mm of shielding structure is presented in Table 20-4. The values in the first row (“Original analysis”) represent the sizing values which were used in the analysis. The values in the second row (“Revised analysis”) came during the IFP and were not able to be injected into the baseline design. As can be seen, the revised analysis predicts a significantly milder radiation environment beyond 4 mm. This is driven by the corrected trajectories (inclined flyby, versus a near-equatorial earlier assumption) for the flyby around Jupiter.

Note that the high predicted radiation doses for the Uranus case are driven by the Jupiter flyby, during which a significant dose is imparted. For the "original analysis" environment, it was also predicted that beyond 10mm Al, the TIDL is driven by the electron environment, which cannot be compensated further by increasing aluminium thickness.

Taking into account the "original analysis" values (predicting 164 krad behind 10 mm Al), 2 options were considered for radiation shielding:

1. Shield sensitive units individually to 10 mm Al (4 mm from the spacecraft structure + 6 mm additional shielding). Any units which still cannot survive the residual TIDL of 164 krad would require delta-design and re-qualification to this level.
2. Shield the entire inner spacecraft (with a radiation "vault") to 10 mm Al. Again, any units which still cannot survive the residual TIDL of 137 krad would require delta-design and re-qualification to this level.

Uranus	**TIDL (krad)**		
	2.5 mm (~100mil)	4 mm	10mm
Original analysis	1350	509	164
Revised analysis	1130	155	40

Table 20-4: Radiation levels for Uranus Orbiter

As stated above, it was assumed that each unit sees an equivalent of 4 mm of Al shielding from the spacecraft structure and surrounding equipment. Table 20-5 presents the additional shielding required by each unit for Option 1, up to a maximum of 6 mm Al (giving 10 mm Al total).

From Table 20-5, and given the TIDS of each unit, at least 7 equipment from the Uranus orbiter would require extra shielding and delta-design / re-qualification. As for the Neptune case, the TIDS of the communication subsystem equipment and payload were identified during the course of the study. In a worst case scenario, they might require up to 150 kg of radiation shielding and delta-design / re-qualification. Thus, these subsystems should be further studied.

Unit	TIDS (krad)	Uranus Option 1			
		Thickness required	Thickness applied (assuming 4 mm provided by structure and unit)	Aluminium Radiation shielding mass (kg)	Delta-design / re-qualification required 164 krad
Gyro	20	>10 mm	6 mm	1.3	Yes
IMU	100	>10 mm	6 mm	2x4.3	Yes
NavCam	2000	None	None	0	No
RW	20	>10 mm	6 mm	4x2.9	Yes
STR	2000	None	None	0	No
CDMU	100	>10 mm	6 mm	2x6.5	Yes
RIUC	100	>10 mm	6 mm	1.9	Yes
PCDU	50	>10 mm	6 mm	6.7	Yes
Batteries	4000	None	None	0	No
Radiation monitor	100	>10 mm	6 mm	1.7	Yes
Comms*	TBD	>10 mm	6 mm	20	TBD
Payload	TBD	>10 mm	6 mm	130	TBD
Total				197	
Total w/ 20% margin				237	

*includes shielding of Ka and X-band EPC, TWT and X-band Transponder

Table 20-5: Uranus orbiter radiation trade-off

As such, for Option 1, approximately 237 kg (including 20% margin) would be required for the known cases. This is mostly driven by the unknown requirements of the payload, which is assumed to require 130 kg shielding.

For Option 2, assuming a spacecraft diameter of 3.5m and height of 2.1m, a total shielding mass of 374 kg (including 20% margin) was estimated. As such, the Option 1 shielding concept was selected for the mission baseline.

Revised analysis

Using the values of the “revised analysis”, the required shielding mass projections would not change. Note however, that this is based on the simplified analysis performed. In this analysis, a simple step-change at 2.5 mm, 4 mm and 10 mm Al was considered. No interpolation was considered between these points. As such, any unit which cannot survive the 4 mm level was automatically given the full 10 mm shielding.

For the “original analysis”, this was less significant as the variation was from 509 krad at 4 mm to 164 krad at 10 mm. No units were identified which had a TIDS within these two data points.

However for the “revised analysis”, the IMU, CDMU, RIUC, PCDU and radiation monitor all had TIDS within the two data points (155 krad at 4 mm, 40 krad at 10 mm). As such, their required shielding would not be the full 10 mm. Such a reduction in the estimate would be the result of an additional analysis.

Note that in the “revised analysis” case, the IMU, CDMU, RIUC, PCDU and radiation monitor would also not require delta-design / re-qualification, if given the correct shielding.

Should the same approach be followed for the Uranus design case as for Neptune, whereby it was assumed that all units would be able to reach a TIDS of minimum 60 krad (excluding RHA margin), then the shielding values via extrapolation would be lower still.

Radiation shielding conclusions

Given the late provision of the “revised analysis” predictions, the system baseline maintains the values from the “original analysis”. Future work should include a reiteration of the design based on the “revised analysis”, including extrapolation for radiation shieldings of varying thickness.

In addition, future work must also address exposed surfaces which may not see significant shielding from the spacecraft structure, such as external surfaces and optical elements. Such elements may still see TIDLs an order of magnitude higher than for the Neptune case. Furthermore, the coupling of the interplanetary transfer properties, e.g. launch date, and the radiation exposure seen during the Jupiter flyby should be investigated further.

20.2.4 Mission System Architecture

20.2.4.1 Mission timeline

The mission timeline is presented in Figure 20-2. Note that the proposed mission foresees two orbiters, one provided by ESA (which journeys to Uranus) and one provided by NASA (which journeys to Neptune).

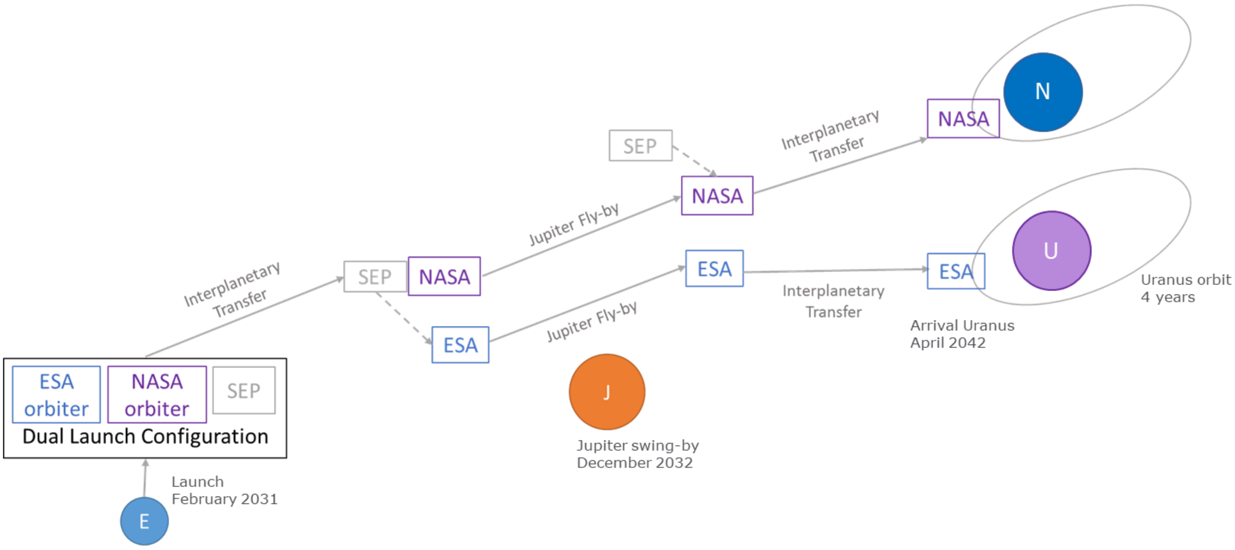

Figure 20-2: Mission timeline (Uranus)

As for the Neptune design case, the two orbiters are to be launched in a dual launch configuration on an SLS Block-1B in February 2031. The NASA orbiter is still assumed to be attached to the Solar Electric Propulsion (SEP) unit, which provides power (to the NASA orbiter) and propulsion (to the stack) up until just after a Jupiter swing-by in December 2032. Shortly before the Jupiter swing-by, the ESA orbiter detaches from the NASA orbiter. It thereafter travels alone to Uranus.

The ESA orbiter should arrive at Uranus in April 2042. Note that this is over two years earlier than in the Neptune design case. The science operations phase at Uranus is envisaged to include planetary science of Uranus, coupled with multiple fly-bys of some of its moons. This tour was not analysed in detail, however a 4 year science operations duration was assumed, based on the initial tour design of RD[1].

20.2.4.2 Mission phases

The mission phases are presented in Table 20-6. The durations until the “independent swing-by phase [of Jupiter]” are the same as for the Neptune design case. As above, the baseline science operations phase duration was set at 4 years. This covered the worst sizing case for the power subsystem (i.e. eMMRTGs) and unit radiation tolerance and reliabilities.

Mission Phase	Duration
(LEOP and) transfer phase [to Jupiter]	1.5 - 2 years
Independent swing-by phase [of Jupiter]	~6 months (TBC)
Cruise phase	9.5 years
Insertion phase	1-2 weeks (TBC)
Science phase	4 years
Disposal	TBD
TOTAL:	**~15.5 years**

Table 20-6: Mission phases (Uranus)

The duration of the insertion phase (for operational constraints) and disposal are also to be clarified in later work.

20.2.4.3 Science operations timeline

The science operations to be performed in Uranus orbit include a mixture between planetary observations and measurements of some Uranian moons (e.g. Titania, Oberon, Umbriel, Miranda and Ariel). As for the Neptune design case, the Uranian moon measurements are performed during close fly-bys, while the majority of the Uranus science is performed at Uranus periapsis. A reference science timeline was defined in order to size the system, as represented in Figure 20-3. Note however that this does not correspond precisely to any single orbit case identified by mission analysis.

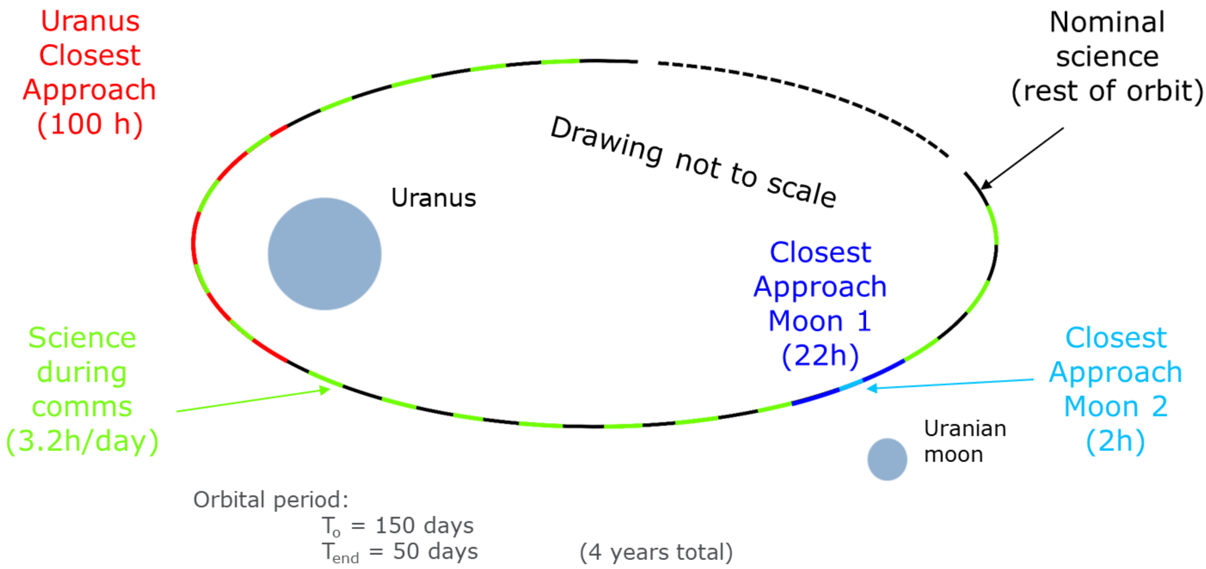

Figure 20-3: Science operations timeline (Uranus)

The reference orbit includes 100 hours of Uranus periapsis science, during the "Uranus Closest Approach" phase. This is broken only intermittently by 3.2 hour communications windows, budgeted for one window per 24 hours.

During the Uranian moon fly-bys, there are two science phases: the "Closest Approach Moon 1" (11 hours before and after the closest point to the moon), and the "Closest Approach Moon 2" (the 2 hours closest to the moon during the fly-by). This covers a total period of 24 hours of moon science. During this phase, there are no communications back to Earth.

For the remainder of the orbit, the orbiter performs "Nominal science". This uses a reduced payload selection, in order to use the surplus power from the RTG's to charge the spacecraft batteries for the higher-consumption phases. During this part of the orbit, the spacecraft also performs 3.2 hours communications of science data per day.

Note that the reference timeline considers a reduction in the orbital period over the 4 years of science performed at Uranus. This reduces from an orbital period of 150 Earth days at the initial orbit, to 50 Earth days by the end of the mission. This was sized on a preliminary understanding of the Triton tour envisaged by NASA in RD[1]. Later analysis revealed that this assumption was incorrect, and that an orbital period reduction of initially 107 Earth days down to 38 Earth days by end-of-mission was more likely. This however could not be re-addressed in the current work. Note that such sizings are highly dependent on the Uranus moon fly-by tour selected, and as such this should be further iterated in future work.

20.2.4.4 System modes

The orbiter system modes are defined in Figure 20-4.

Launch and early operationa mode [LEOP]	• Commissioning of the equipment, while still attached to the SEP stage.
Transfer mode [TM]	• Orbiter is still connected to the SEP stage Only few equipment are turned on periodically for readout or possible checks
Cruise mode [CM]	• After Jupiter fly-by and until arrival to the planet • No instruments turned on
Manoeuvre mode [MM]	• Performing main orbit manoeuvres using the main thruster(s).
Observation mode [OBM]	• Most payload instruments operate
Communication mode [COMM]	• Communication back to Earth and radio science
Nominal Science Mode [NSM]	• A reduced set of payload instruments operate

Figure 20-4: System modes (Uranus)

20.2.4.5 Science sub-modes

The science sub-modes are analogous to those for the Neptune orbiter.

20.2.4.6 Mission options

RD[1] also includes a science operations phase of 2 years at Uranus. As discussed above, the duration of the science phase is highly dependent on the moon tour selected, and it's compatibility with the science goals of the mission. Reducing the science phase to 2 years would improve the eMMRTG output power at EOM (due to short mission duration), while also reducing the requirements on unit radiation tolerance and reliabilities, delta-V on-orbit, and mission cost. As such, it would provide an attractive alternative, as long as the science requirements can be fulfilled.

20.2.5 System Baseline Design

The baseline orbiter design is summarised in Table 20-7.

Orbiter	
Mass (Incl 20% system margin)	Dry mass: 1914 kg Propellant mass (excl. margin): 2484 kg Wet mass: 4398 kg
Payload	Camera Imaging Spectrometer Ion and Neutral Mass Spectrometer Magnetometer Microwave radiometer Ultra Stable Oscillator (USO) Ka-band transponder
Propulsion	1x main bipropellant thruster (1000 N) 16x RCS thrusters (10 N) 3x pressurant tanks (120 L) 4x propellant tanks (550 L)
AOGNC	1x coarse rate sensor 2x navigation cameras 2x IMUs 2x star trackers 4x reaction wheels (+ RCS thrusters)
Communications	X-band uplink/downlink Ka-band downlink (94 kbps) Science volume downlinked: 1.09 Gb/day Communication window duration: 3.2 h/day Data volume generated by EOM: 1.6 Tb
Power	3x eMMRTGs (EOM Power = 90W for 4 year science phase) 4x 48kg batteries
Data Handling	Redundant OBC + 1Tbit of storage
Structures	303 kg
Thermal	Heaters + use of the eMMRTG thermal dissipation

Table 20-7: Orbiter system baseline (Uranus)

20.2.6 System Budgets

20.2.6.1 Mass budget

The mass budget for the Uranus orbiter is presented in Table 20-8. Residuals' propellant mass margins are already included in the values provided by chemical propulsion (see Chemical Propulsion Chapter).

SC Mass Budget		Mass [kg]
Attitude, Orbit, Guidance, Navigation Control		60.40
Communications		71.64
Chemical Propulsion		233.94
Data-Handling		38.48
Instruments		118.41
Mechanisms		39.00
Power		350.04
Structures		303.26
Radiation Shielding		237.02
Thermal Control		65.89
Radiation Instrumentation		1.49
Harness	5%	75.90
Dry Mass w/o System Margin		**1595.48**
System Margin	20%	319.10
Dry Mass incl. System Margin		**1914.58**
CPROP Fuel Mass		931.72
CPROP Fuel Margin	0%	0.00
CPROP Oxidizer Mass		1537.34
CPROP Oxidizer Margin	0%	0.00
CPROP Pressurant Mass		14.70
CPROP Pressurant Margin	0%	0.00
Total Wet Mass		**4398.33**

Table 20-8: Uranus orbiter mass budget

The corresponding equipment list is presented in Table 20-9.

Equipment	#	Mass (kg)	Total Mass (kg)	Mass margin (%)	Total mass incl. margin (kg)
SC (Spacecraft)					
AOGNC			**56**	**7.86**	**60.4**
IMU_Astrix_1090A_1 (IMU Airbus Astrix 1090A #1)	2	5.00	10.00	5	10.5
NavCam_1 (NavCam #1)	2	11.00	22.00	5	23.1
RW_HR04_1 (RW Honeywell HR04 #1)	4	2.60	10.40	20	12.5
STR_HydraEU_Juice_1 (STR Sodern Hydra JUICE Electronics Unit #1)	2	3.60	7.20	5	7.6

Equipment	#	Mass (kg)	Total Mass (kg)	Mass margin (%)	Total mass incl. margin (kg)
STR_HydraOH_Juice_1 (STR Sodern Hydra JUICE Optical Head #1)	2	2.80	5.60	5	5.9
GYRO_Sireus (GYRO Selex Galileo Sireus)	1	0.80	0.80	10	0.9
COM			**64.20**	**11.59**	**71.6**
HGA (High Gain Antenna)	1	33.00	33.00	10	36.3
KaEPC (Ka-Band Electronic Power Conditioning)	2	1.30	2.60	20	3.1
KaTWT (Ka-Band Traveling Wave Tube)	2	0.80	1.60	20	1.9
LGA_LHCP (Low Gain Antenna - LHCP)	1	0.90	0.90	5	0.9
LGA_RHCP (Low Gain Antenna - RHCP)	1	0.90	0.90	5	0.9
RFDN (Radio Frequency Distribution Network)	1	13.00	13.00	20	15.6
XEPC (X-Band Electronic Power Conditioning)	2	1.30	2.60	5	2.7
XKaXPND (X/X/Ka-Band Transponder)	2	4.00	8.00	5	8.4
XTWT (X-Band Traveling Wave Tube)	2	0.80	1.60	5	1.7
DH			**32.07**	**20**	**38.5**
RIUC (Remote Inteface Unit Centralised)	1	8.33	8.33	20	10.0
RIUD (Remote Interface Unit Decentralised)	1	7.08	7.08	20	8.5
CDMU_1 (Computer and Data Management Unit #1)	2	8.33	16.66	20	20.0
INS			**98.94**	**19.68**	**118.4**
Cam (Camera)	1	16.00	16.00	20	19.2
Im_spec (Imaging Spectrometer)	1	15.50	15.50	20	18.6
INMS (Ion and Neutral Mass Spectreometer)	3	12.00	36.00	20	43.2
Mag (Magnetometer)	1	4.56	4.56	20	5.5
Micro_rad (Microwave radiometer)	1	19.34	19.34	20	23.2
USO (Ultra Stable Oscillator)	1	2.00	2.00	20	2.4
Ins_KaEPC (Instrument Ka-Band Electronic Power Conditioning)*	1	1.30	1.30	5	1.4
InsKaTWT (Instrument Ka Band Traveling Wave Tube)*	1	0.80	0.80	5	0.8
Ka_Transp (Ka-band Trransponder)	1	3.44	3.44	20	4.1
MEC			**35.00**	**11.43**	**39.0**
magBOOM (Deployable magnetometer boom)	1	30.00	30.00	10	33.0
SEP_separation (SEP stage separation [SC side])	1	5.00	5.00	20	6.0
PWR			**324.90**	**7.74**	**350.0**
Bat_Orb (Battery_Orbiter)	4	43.90	175.60	5	184.4
EMMRTG (Enhanced_Multi_Mission_RTG)	3	45.00	135.00	10	148.5

Equipment	#	Mass (kg)	Total Mass (kg)	Mass margin (%)	Total mass incl. margin (kg)
PCDU_Orb (Power Conditioning & Distribution Unit_Orbiter)	1	10.30	10.30	20	12.4
Ext_Pwr_Shnt (External power shunt)	1	1.00	1.00	20	1.2
Res_Pwr_Shnt (Resisitive power shunt)	3	1.00	3.00	20	3.6
STR			**252.72**	**20**	**303.3**
APs (Assembly Panels)	1	54.82	54.82	20	65.8
BP (Bottom Panel)	1	18.16	18.16	20	21.8
CPROP_TD (CPROP_Tank Deck)	1	21.88	21.88	20	26.3
MC (Module Collars)	1	22.00	22.00	20	26.4
SPs (Shear_Panels)	1	28.64	28.64	20	34.4
TP (Top Panel)	1	18.16	18.16	20	21.8
TR (Tube Rings)	1	12.74	12.74	20	15.3
TSS (Tank Supporting Struts)	1	66.00	66.00	20	79.2
TST (Tank Supporting Tube)	1	10.32	10.32	20	12.4
TC			**59.72**	**10.34**	**65.9**
TCS (Thermal Control Subsystem)	1		0.00	0	0.0
NO_BP (NO_Black_Paint)	1	10.00	10.00	10	11.0
NO_Louvre (NO_Louvres)	1	2.05	2.05	20	2.5
NO_MLI_ex (NO_MLI_external_22-layer)	1	32.00	32.00	10	35.2
NO_MLI_HGA (NO_MLI_HGA_10-layer)	1	1.60	1.60	10	1.8
NO_MLI_int (NO_MLI_internal_10-layer)	1	3.20	3.20	10	3.5
NO_Rad (NO_Radiator_SSM-tape)	1	0.20	0.20	10	0.2
NO_WP (NO_White_Paint)	1	0.80	0.80	10	0.9
NO_MLI_RTG_rad (NO_MLI_RTG_radiative_shield)	1	0.80	0.80	10	0.9
NO_MLI_RTG_ShuntRad (NO_RTG_ShuntRadiator)	1	1.88	1.88	10	2.1
NO_Therm_01 (NO_Thermistor)	40	0.06	2.40	10	2.6
O_Heater_01 (O_Heater)	80	0.06	4.80	10	5.3
CPROP			**221.6**	**5.56**	**233.9**
Biprop_FDV_1 (Biprop_FillDrain_Valve)	9	0.07	0.63	5	0.7
Biprop_Filter_1 (Biprop_Filter)	4	0.08	0.31	5	0.3
Biprop_LP_Trans_1 (Biprop_LP_Transducer)	4	0.22	0.86	5	0.9
Biprop_LV_1 (Biprop_Latch_Valve)	4	0.75	3.00	5	3.2
Biprop_NRV_1 (Biprop_Non_Return_Valve)	4	0.59	2.34	5	2.5
Biprop_Pipes (Biprop_Pipes)	1	8.00	8.00	20	9.6
Biprop_Thruster_Main_1 (Biprop_Thruster_Main #1)	1	7.80	7.80	5	8.19
Biprop_PR_1 (Biprop_PressureRegulator)	2	1.00	2.00	5	2.1
Biprop_Pres_Tank_1	3	23.50	70.5	5	7.40

Equipment	#	Mass (kg)	Total Mass (kg)	Mass margin (%)	Total mass incl. margin (kg)
(Biprop_Pressurant_Tank)					
Biprop_Prop_Tank_1 (Biprop_Prop_Tank)	4	28.29	113.18	5	118.84
Biprop_SMA_Valve_1 (Biprop_SMA_Valve)	2	0.16	0.32	20	0.4
Biprop_Thruster_RCS_1_01 (Biprop_Thruster_RCS #1)	16	0.65	10.40	5	10.9
Biprop_HP_LV (Biprop_HP_Latch_Valve)	1	0.80	0.80	5	0.8
Biprop_HP_Trans (Biprop_HP_Transducer)	1	0.22	0.22	5	0.2
Biprop_PV_1 (Biprop_Pyro_Valve)	4	0.32	1.26	5	1.3
RAD			**1.35**	**10**	**1.5**
rad_mon_ngrm (Radiation Monitor NGRM)	1	1.35	1.35	10	1.5

****These equipment are here modelled as part of the instruments (payload) as they are only required to perform radio science, but are actually integrated into the architecture of the communication subsystem***

Table 20-9: Uranus orbiter equipment list

20.2.6.2 Power budget

The Uranus orbiter power budget was assumed the same as the Neptune orbiter power budget. Although the later has an additional main thruster, which is activated during the manoeuvre mode, both systems were designed considering only one thruster in terms of power consumption. This should however not affect the power subsystem design which is driven by the science operations.

20.2.6.3 Data budget

The Uranus orbiter data budget is the same as the Neptune orbiter data budget.

20.2.6.4 Dissipation budget

The Uranus orbiter dissipation budget was assumed the same as the Neptune orbiter dissipation budget.

20.2.6.5 Delta V budget

The Uranus orbiter Delta V budget is presented in Table 20-10.

Delta-v Budget	Manoeuvre type	Orbiter to Uranus	Unit	Comment
Jupiter fly-by Targeting	stochastic	15	m/s	
Orbital Insertion	deterministic	1878	m/s	From propulsion, consedering baseline T/M ratio
Triton/Uranus Moon Targeting	deterministic	201	m/s	
Planet Tour Deterministic	deterministic	85	m/s	
Planet Tour Stochastic	stochastic	30	m/s	
Planet Tour Future Design	deterministic	20	m/s	
Margin on stochastic delta-		0	%	3sigma values, no margin

Delta-v Budget	Manoeuvre type	Orbiter to Uranus	Unit	Comment
v				applied
Margin on deterministic delta-v		5	%	
Total det. and stoch. Manoeuvres		2338.2	m/s	
Disposal manoeuvre		10	m/s	
Margin on disposal manoeuvre		0	%	
Total disposal manoeuvre		10	m/s	
AOCS delta-v		149.3	m/s	Margin on total propellant estimated by propulsion
Margin on AOCS delta-v		0	%	
Total AOCS delta-v		149.3	m/s	
Total delta-v without margin		**2388.3**	**m/s**	
Total delta-v including margin		**2497.5**	**m/s**	

Table 20-10: Uranus orbiter Delta V budget

20.2.7 System Options

20.2.7.1 Payload timeline (JV)

The assumptions for the payload timeline and the communication window duration are the same as for the Neptune case. With the exception of the downlink data rate which is 94 kbps in the Uranus case.

20.2.7.1.1 Results

The results for the communication window duration have been sized for the 50 day orbit, to ensure that the total generated energy equals the total consumed energy per orbit.

The total communication window duration in this case is 3.2 hours. Table 20-11 shows that the worst case is the 50 day orbit and that there is energy available in case of a different orbit duration. Table 20-12 shows that, in contrast to the Neptune case, there is a positive data margin for all of the orbits.

	Energy			
Days	Generated	Consumed	Margin	
50	291600	291600	0	Wh
75	437400	425107	12293	Wh
100	583200	558614	24586	Wh

Table 20-11: Power consumed in each science mode for the instruments and the platform

Energy				
Days	Generated	downlinked	Margin	
50	32	55	24	Gb
75	40	83	44	Gb
100	47	111	64	Gb

Table 20-12: The total data generate, downlink availability and margin for a 50, 75 and 100 day orbit

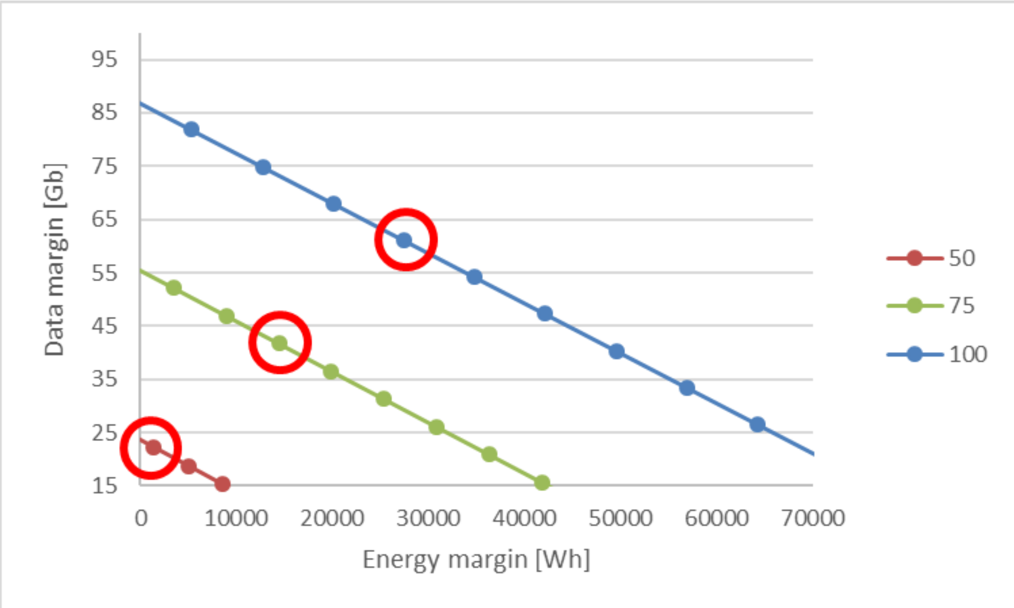

Figure 20-5: The energy margin available plotted against the data margin available. The circled points are the data points from Table 20-11 and Table 20-12

The numbers in the following pictures are preliminary numbers that will change significantly with minor changes in the power budget.

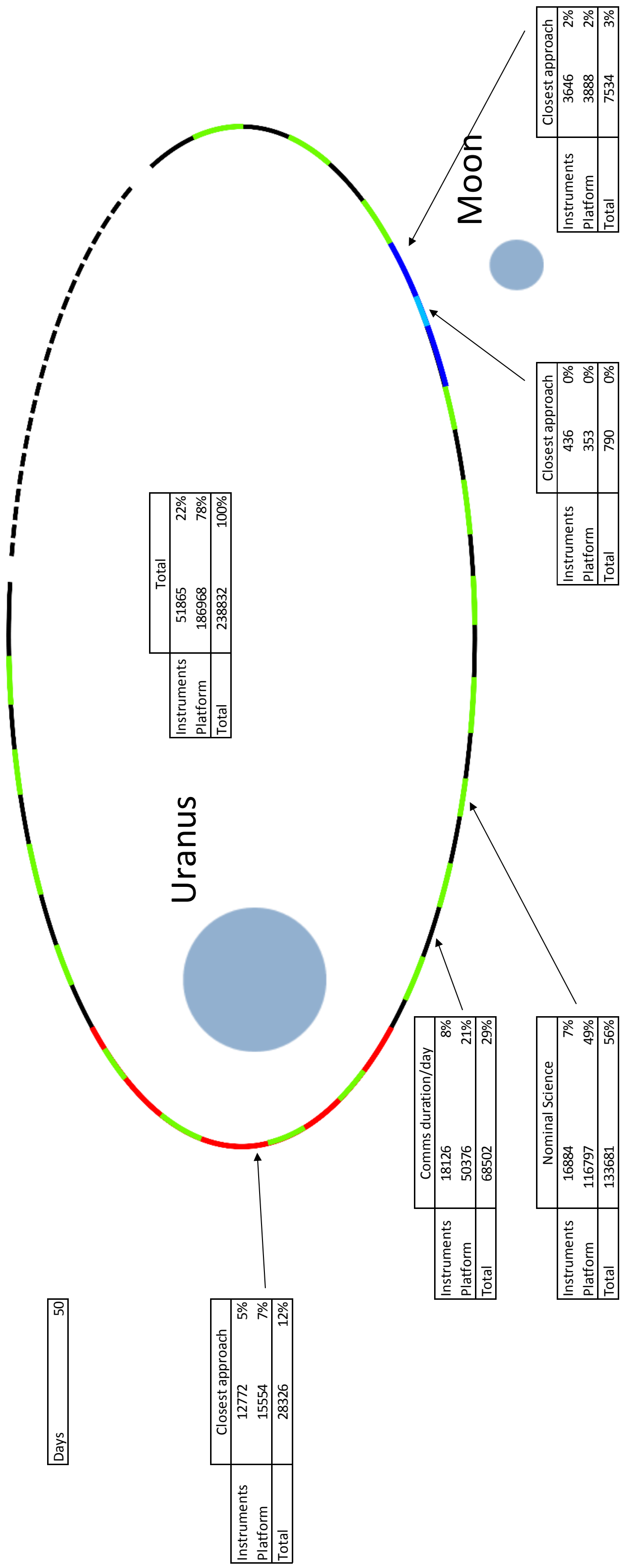

Figure 20-6: The consumed energy per mode and per orbit in kW and percentage over the total orbit for the 50 day orbit

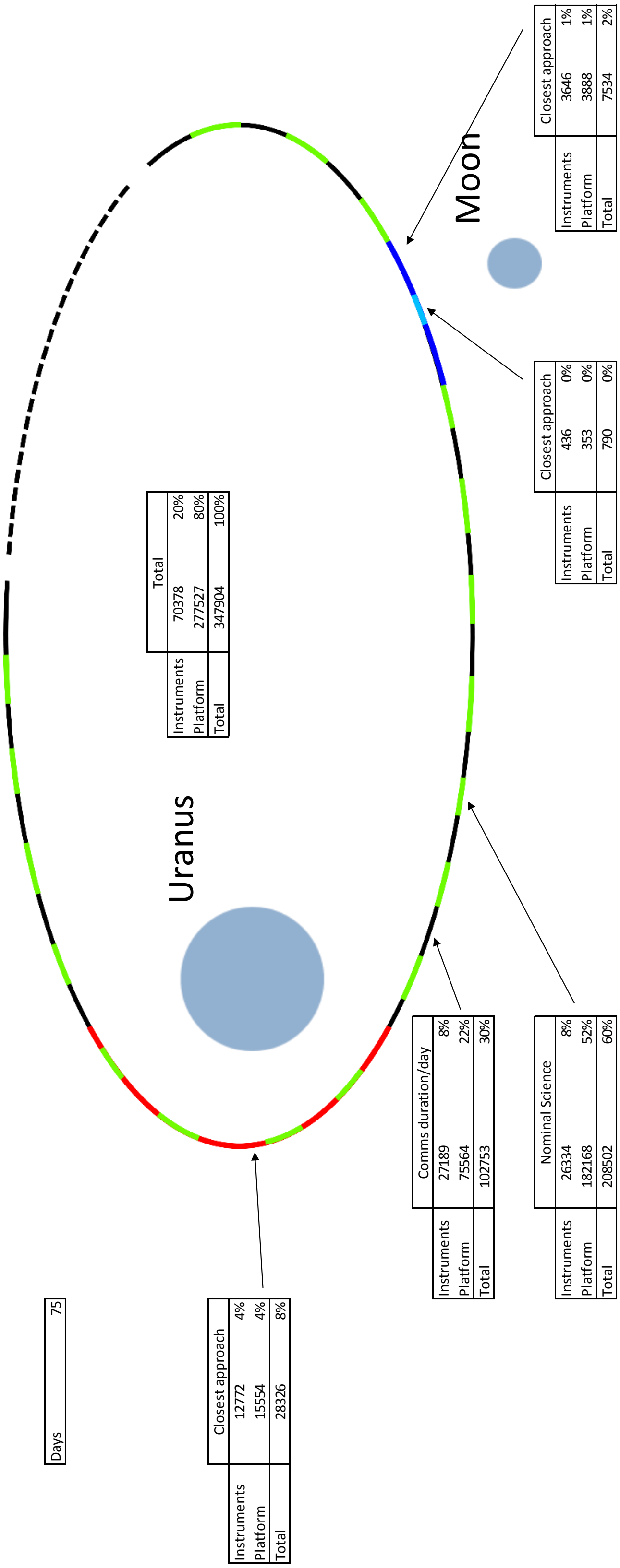

Figure 20-7: The consumed energy per mode and per orbit in kW and percentage over the total orbit for the 75 day orbit

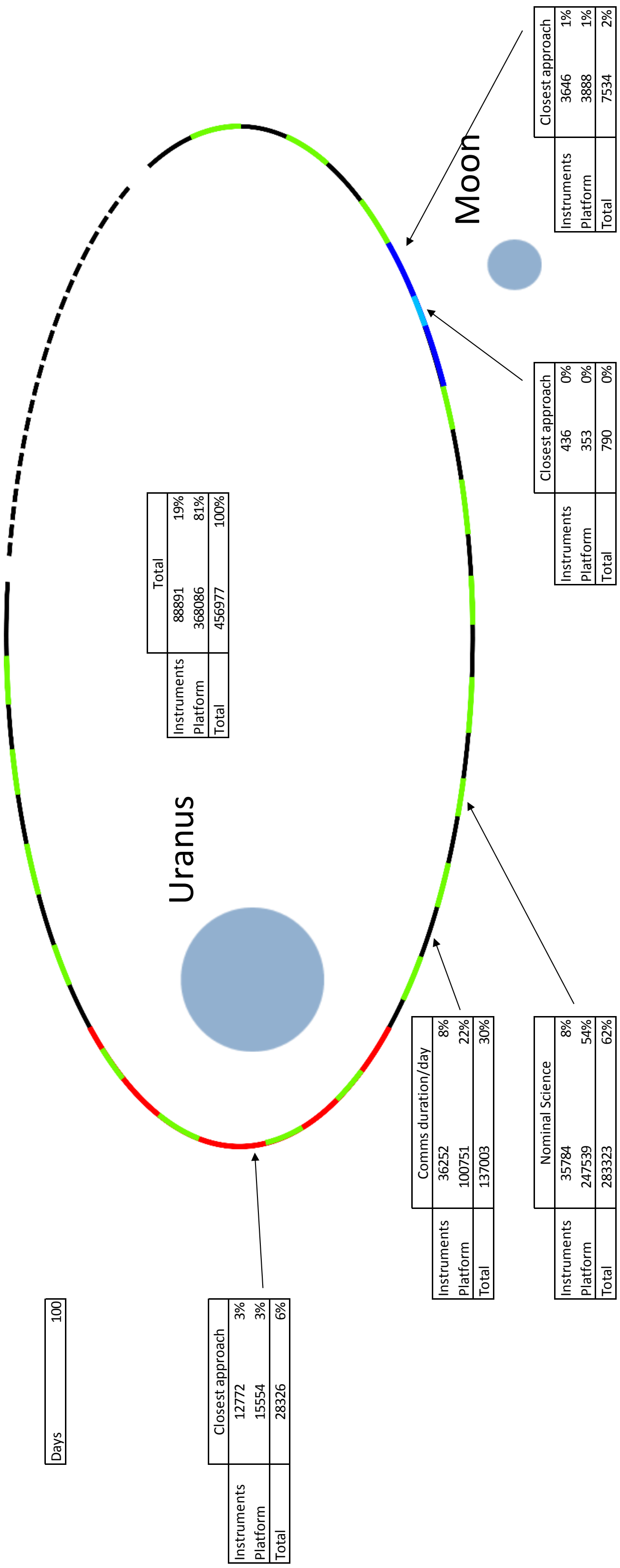

Figure 20-8: The consumed energy per mode and per orbit in kW and percentage over the total orbit for the 100 day orbit

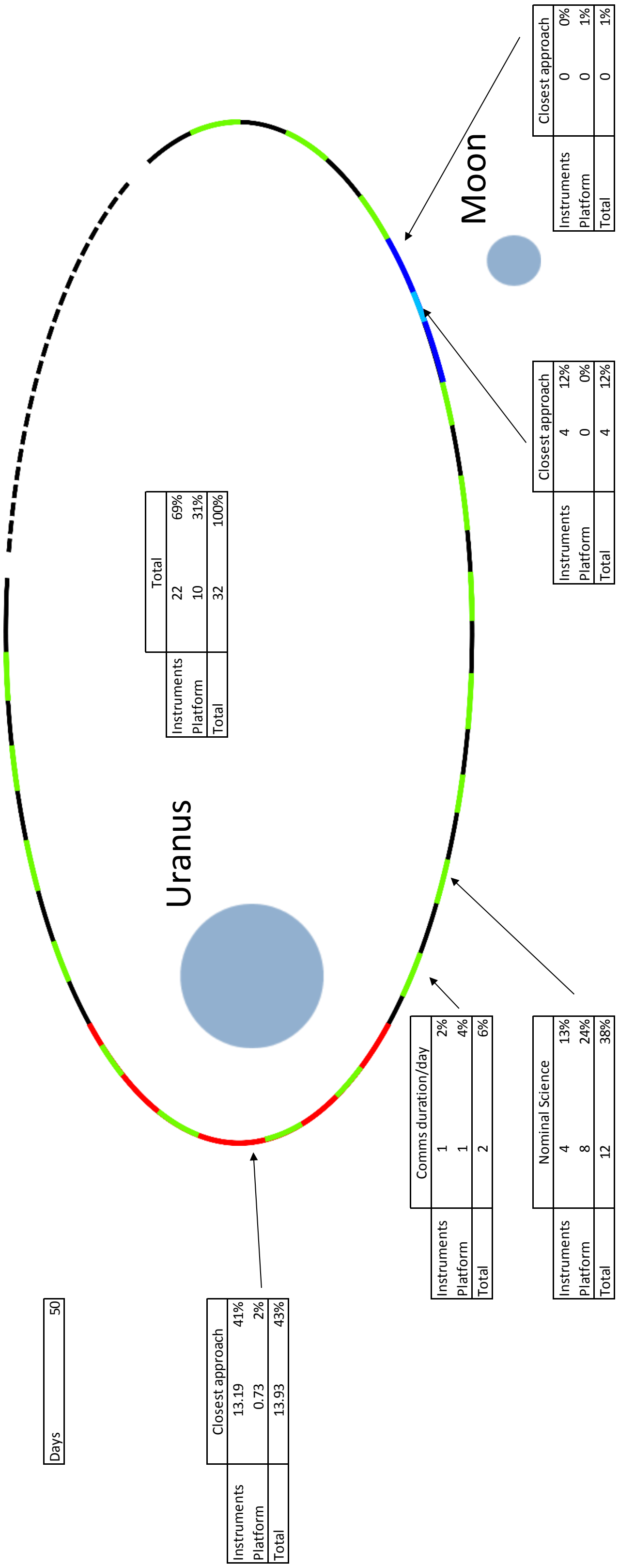

Figure 20-9: The generated data per mode in Gb and percentage over the total orbit for the 50 day orbit

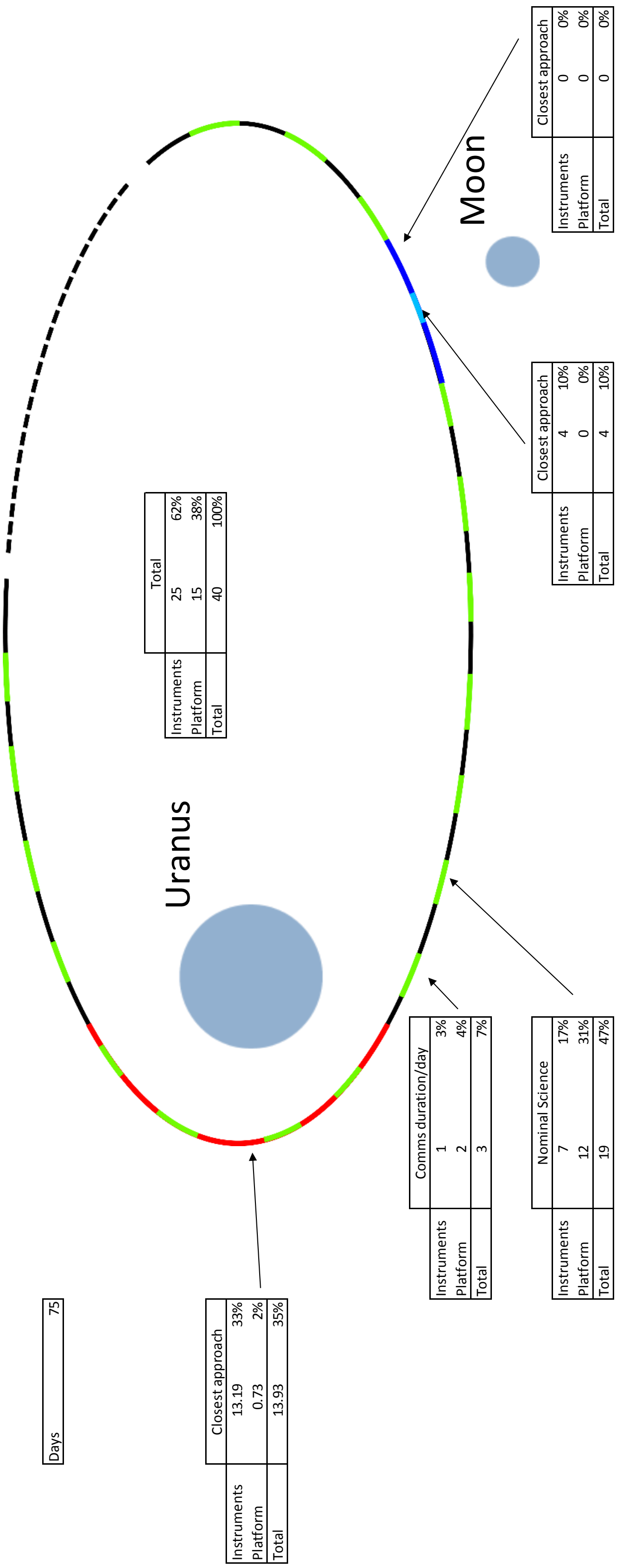

Figure 20-10: The generated data per mode in Gb and percentage over the total orbit for the 75 day orbit

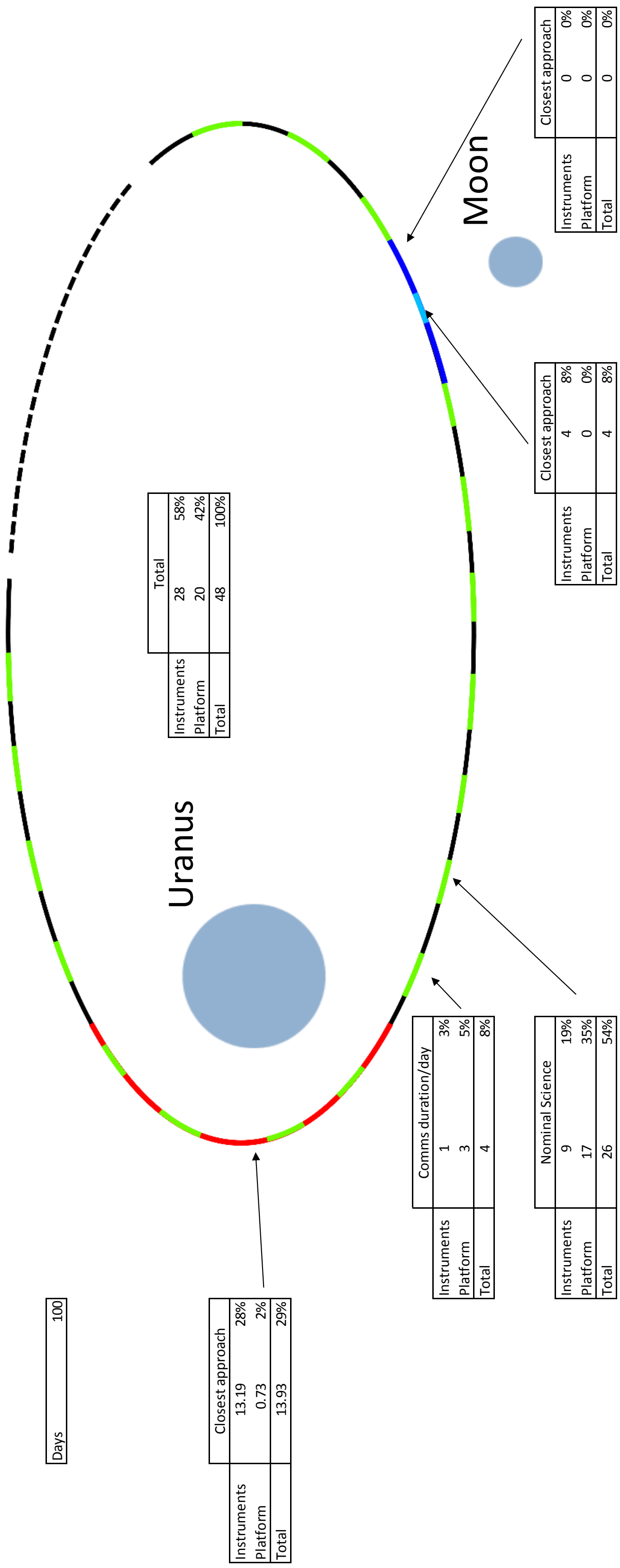

Figure 20-11: The generated data per mode in Gb and percentage over the total orbit for the 100 day orbit

20.2.7.2 Number of RTGs

An analysis has been performed to see if it is feasible to downlink all of the generated data using only 2 RTGs.

However, looking at the total power in the spacecraft lowest power mode, 125 W is required for the platform and 18 W for the payload. Adding the 20% system margin, this gives a total power consumption of 172 W (excluding the 90% charging efficiency, which would lead to 191 W at input). Since the 2 RTGs only generate 180W combined, a 2 RTG system would not be feasible even for low power mode of the spacecraft.

20.2.8 Future Work

As for the Neptune case, with the addition of:

- **Further optimisation of orbiter design for Uranus:**
  As discussed in Section 20.2.2, the study objectives foresaw maximum reuse of the design between the Neptune and Uranus design cases. As such, some subsystems (in particular the communications subsystem) may be oversized for the design case. Future work should address a detailed and optimised design for the Uranus case.

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21 URANUS PAYLOAD

21.1 Atmospheric Probe

For the Uranus mission the same Model Payload was assumed as for the Neptune mission (see Section 6.1.3 and RD[7]).

These instruments are a representation of a possible future payload but are not meant to be understood as a pre-selection of instruments for a potential future mission.

21.1.1 Requirements and Design Drivers

The main design drivers for the Uranus Atmospheric Probe were the same as for the Neptune Atmospheric Probe. The transfer to Uranus is ca. 2 years shorter than to Neptune but the same requirements for instruments temperature and power were assumed for this CDF study.

21.1.2 Assumptions and Trade-Offs

For the Atmospheric Probe Model Payload no trade-offs were performed, as the same instruments were used as identified in the PEP (V) CDF study (RD[7]).

The assumptions that were taken are listed in Table 21-1:

Assumptions	
1	No radiation shielding is assumed. Heating for the instruments shall be provided by RHUs [[add system ref.]].

Table 21-1: Assumptions for the Uranus Atmospheric Probe Model Payload.

For this CDF study no radiation shielding was assumed, but with a calculated TID (Total Ionising Dose) for the Uranus case of ~155 krad with 4 mm Al shielding [[add ref to radiation section]] a suitable radiation shielding approach should be adopted in potential future studies.

21.1.3 List of Equipment

Not applicable.

21.1.4 Options

No Options were studied.

21.1.5 Technology Needs

No new technologies were identified for the Model Payload (see Section 6.1.3, Table 6-3).

21.2 Orbiter

21.2.1 Requirements and Design Drivers

For the Uranus mission the same Model Payload was assumed as for the Neptune mission (see Section 6.2.3.1).

These instruments are a representation of a possible future payload but are not meant to be understood as a pre-selection of instruments for a potential future mission.

The same assumptions were taken for the Orbiter element as for the Neptune case and an adequate shielding strategy shall be devised in a potential future study.

21.2.2 List of Equipment

Not applicable.

21.2.3 Options

No Options were studied.

21.2.4 Technology Needs

No new technologies were identified for the Model Payload.

22 URANUS CONFIGURATION

22.1 Atmospheric Probe

The probe configuration for Uranus remains unchanged; nothing has been added nor changed.

22.2 Orbiter

*/In principle, the orbiter configuration for Uranus remains unchanged from the Neptune design. The propulsion subsystem changed only in terms of number of thruster used, from two main engines to one, and it also uses three helium tanks of the same type, from the family PFVG-120 (see Chapter 25.1.3).

Since the radiation environment is higher than the equipment can tolerate, a shielding structure needs to be employed in order to protect them.

23 URANUS STRUCTURE

23.1 Atmospheric Probe

The structure of the Uranus Probe is assumed to be identical to the probe design for Neptune.

23.2 Orbiter

The structure of the Uranus Orbiter is assumed to be identical to the Orbiter design for Neptune.

24 URANUS MECHANISMS

24.1 Atmospheric Probe

24.1.1 Requirements and Design Drivers

Mechanism requirements and design drivers are equal to the ones for the Neptune mission.

24.1.2 Baseline Design

The design of the probe mechanisms is the same as for the Neptune mission.

24.2 Orbiter

24.2.1 Requirements and Design Drivers

Mechanism requirements and design drivers are equal to the ones for the Neptune mission.

24.2.2 Baseline Design

The design of the orbiter mechanisms is the same as for the Neptune mission.

25 URANUS PROPULSION

25.1 Orbiter

25.1.1 Requirements and Design Drivers

The requirements for the propulsion system are derived from the main requirements for the delta v and the mission and the ground operations.

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
PROP-010	Propulsion system provides necessary thrust and delta v for the mission manoeuvres	
PROP -020	Propulsion system provides torques to compensate the main thruster misalignments and for all other AOCS manoeuvres	
PROP -030	Propulsion system has at least three barriers for safety reasons on ground	
PROP -040	Propulsion system includes the measurement of the pressures within the subsystem at mandatory locations	
PROP -050	Propulsion system provides means to isolate potential mechanical pressure regulator leakage through the mission	
PROP -060	Propulsion system provides means to isolate the main engine in case of major leakage	
PROP -070	Propulsion system incorporates per branch a serial redundant pressure regulator	
PROP -080	Propulsion system includes Fill and drain valves for filling and testing of the propulsion system on ground	

25.1.2 Assumptions and Trade-Offs

The following table includes the assumptions used during the mission scenario.

Assumptions	
1	Gravity losses are linearly interpolated from the data provided by Mission Analysis. Additionally, a margin of 5% was taken for the delta v demands of the gravity losses.
2	The AOCS mass was modelled by using 5% of the total propellant mass used during the mission for the delta v manoeuvres. This propellant mass is split into 3% after the main Uranus Orbit insertion manoeuvre and 2 % after the mission.
3	For the bipropellant system, a mixture ratio of 1.65 was assumed. The dual mode systems used a mixture ratio of 1.43
4	No redundancy need for the main engine was assumed. It was furthermore assumed that the main engine can be accommodated in such a way to minimise the propellant need for any misalignment or centre of mass shift.
5	The thrust and the specific impulse of the engine was modelled by using the parametric model provided by the supplier.

Assumptions	
6	The pressure set point in the tanks and the orifice of the main engine corresponds to the nominal thrust point of 1kN for the engine, for the case of the mechanical pressure regulator. For the electronic pressure regulator case, adjustment to a thrust of 1.1kN is assumed to be possible.
7	The propellant mass includes a 2% residual mass of the propellant at the end of the mission.
8	The volumes of the tanks are calculated to fulfil the volume margin requirement of around 10%.
9	Uranus orbit insertion burn was done in one single manoeuvre. Tank depletion and corresponding temperature drop is assumed to be isentropic. Temperature threshold for the design and the helium tank size was around -18°C.
10	The tank sizes are derived from the E3000 tank family. Tank heights and masses are using linear interpolation.

For all calculations, the following manoeuvre approach was used to estimate the propellant budget:

Manoeuvre	velocity increment [m/s]	propellant mass [kg]
Orbiter Deflection Manoeuvre	11.2	
UOI	Delta v with gravity losses in comparison to thrust of propulsion system	
AOCS Mass		Propellant percentage of 3% for all delta v manoeuvres
Target Manoeuvre	211.1	
Tour	85.0	
Tour	30.0	
Tour Margin	31.5	
Disposal	10	
AOCS mass		Propellant percentage of 2% for all delta v manoeuvres

Table 25-1: Delta v and propellant mass assumption for the Uranus Orbiter case

25.1.3 Baseline Design

The baseline design of the Uranus Orbiter case is in principle the same as the one for the Neptune Orbiter. The differences are in the underlying requirements in terms of delta v and gravity loss dependence on thrust, which in this case led to a system using one engine only.

One particular point to mention is the radiation impact on the equipment. Since the level of radiation was indicated to be around 155 krad, some of the equipment currently in development (like the LEROS-4 engine) needs detailed assessment. For some of the equipment, special radiation shields can be ordered as well (one example being the SAPT). This extra equipment was currently not baselined and further analysis would need to be done to investigate this in more detail.

25.1.4 List of Equipment

The following list includes the equipment and masses for the baseline concept:

Description	Type	Amount	Mass per unit	Margin	Mass incl. margin
Pipes	Pipes	1	8	0.2	9.6
AOCS Engines	S10-18	16	0.65	0.05	10.92
Main Engine	LEROS-4B	1	7.8	0.05	8.19
Fuel Tank	E3000	2	28.29	0.05	59.42
Oxid Tank	E3000	2	28.29	0.05	59.42
Fill / Drain Valves		9	0.07	0.05	0.6615
LP Pressure Transducer	SAPT	4	0.216	0.05	0.9072
HP Pressure Transducer	SAPT	1	0.216	0.05	0.2268
Latch Valve		4	0.75	0.05	3.15
Propellant Filter	RA04822A	4	0.077	0.05	0.3234
Check valve	VN005-001	4	0.585	0.05	2.457
Helium Tank	PVG-120	3	23.5	0.05	74.025
Helium Tank	PVG-65	0	12	0.05	0
Pressure regulator	VACCO	2	1	0.05	2.1
Pyrovalve	Cobham	4	0.315	0.05	1.323
SMA valve	Arianegroup	2	0.16	0.2	0.384
High pressure latch valve	Vacco V1E10560-01	1	0.8	0.05	0.84
Total					233.94

Table 25-2: Propulsion system mass budget

Table 25-2 includes one main engine and the adapted tank size for the updated propellant budget. Table 25-3 lists the propellant masses for each manoeuvre:

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]
Orbiter Deflection Manoeuvre	4400.3	4379.4	15.0	20.90
UOI	4379.4	2341.7	1971.9	2037.67
AOCS Mass	2341.7	2272.5	84.4	69.22
NOI clean-up	2272.5	2272.5	0.0	0.00
Tour Targeting	2272.5	2125.3	211.1	147.28
Tour	2125.3	2065.9	89.3	59.37
Tour	2065.9	2046.3	30.0	19.58
Tour Margin	2046.3	2031.1	21.0	15.21
Disposal	2031.1	2023.9	10	7.20
AOCS mass	2023.9	1977.7	64.91	46.14
Final/Total (Including Residuals)	1914.58		2497.5	2469.06

Table 25-3: Manoeuvre propellant budget

25.1.5 Options

The options investigated included the baseline with an electronic pressure regulator as well. The results are listed in Table 25-4, showing the nearly negligible impact of the thrust on the gravity losses. The dry mass increase of around 15kg is therefore leading to a total amount of extra propellant needed in the order of 38kg.

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]
Orbiter Deflection Manoeuvre	4451.8	4436.0	11.2	15.86
UOI	4436.0	2366.5	1971.8	2069.45
AOCS Mass	2366.5	2296.3	84.7	70.25
NOI clean-up	2296.3	2296.3	0.0	0.00
Triton Target Manoeuvre	2296.3	2147.5	211.1	148.82
Tour	2147.5	2090.1	85.0	57.39
Tour	2090.1	2070.2	30.0	19.89
Tour Margin	2070.2	2047.2	31.5	23.04
Disposal	2047.2	2039.9	10	7.26
AOCS mass	2039.9	1993.1	65.37051861	46.83
Final/Total (Including Residuals)	1929.16		2500.6	2507.97

Table 25-4: Propulsion system with electronic pressure regulator

Additionally, the Uranus case was also investigated without the shielding mass of 237.02kg. This leads to a decrease of the dry mass to 1578.87kg. The propulsion system was changed to the usage of 400l tanks for MMH and MON. The Helium tank sizes also changed back to the original Neptune case of having 2 times 120l tanks and 1 time 65l tank. Using the dry mass as mentioned above leads to the following results in terms of velocity increment and propellant mass consumption:

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]	propellant mass [kg]
Orbiter Deflection Manoeuvre	3618.9	3606.0	11.2	12.84
UOI	3606.0	1937.6	1956.5	1668.40
AOCS Mass	1937.6	1881.1	83.3	56.73
NOI clean-up	1881.1	1881.1	0.0	0.00
Triton Target Manoeuvre	1881.1	1759.2	211.1	121.91
Tour	1759.2	1712.4	85.0	46.84
Tour	1712.4	1696.1	30.0	16.23
Tour Margin	1696.1	1677.2	31.5	18.88
Disposal	1677.2	1671.3	10	5.95
AOCS mass	1671.3	1633.5	64.42	37.82
Final/Total (Including Residuals)	1578.87		2483.0	2025.32

Table 25-5: Manoeuvre propellant budget without shielding mass

25.1.6 Technology Requirements

The technology requirements for the Uranus case kept the same as the ones already mentioned for the Uranus case, no additional needs are identified.

26 URANUS AOCS

26.1 Orbiter

26.1.1 Requirements and Design Drivers

See Neptune orbiter attitude control system chapter. Requirements are assumed unchanged for Uranus, with the exception that Triton is potentially substituted for a Uranus moon or moons.

26.1.2 Assumptions and Trade-Offs

See Neptune orbiter attitude control system chapter.

Uranus has a shorter cruise duration, roughly 2 years less than Neptune.

26.1.2.1 Orbit and fly-by trajectories

Several orbits and fly-by trajectories were considered for the Uranus orbiter mission:

5. Cruise: Heliocentric orbit with periapsis at Jupiter and apoapsis at Uranus
6. Post-probe-injection (if necessary): Low-periapsis Uranus orbit (2000 km altitude periapsis, 225 x R_{uranus} apoapsis)
7. Science: High-periapsis Uranus orbit (20000 km altitude periapsis, 225 x R_{uranus} apoapsis)
8. Science: Uranus moon fly-by (during high-periapsis Uranus orbit) at 100 km altitude and relative speed of 3.9 km/s.

26.1.3 Baseline Design

See Neptune orbiter attitude control system chapter for main architectural design, etc.

26.1.3.1 Sensor selection

There are two sensors that may require parts upgrade and/or local/global shielding to handle the higher dosage of the Uranus mission, due to the closer approach to Jupiter during gravity assist. These are described as follows:

26.1.3.1.1 Coarse Rate Sensor

TAS SiRUES radiation susceptibility information not available, but assumed to be < 100 krad. If parts upgrade or shielding are not feasible options or deemed to costly or massive, a third Astrix 1090 gyro (IMU without accelerometer) could be embarked in its place. This backup gyro would only be used in safe mode. The disadvantage would be that the safe mode sensor hardware type would then be identical to that used in nominal operations and may be more costly.

26.1.3.1.2 IMU

Airbus Astrix 1090A radiation tolerance is 100 krad. Some additional shielding may be required.

26.1.3.2 Actuator selection

26.1.3.2.1 Environmental torque disturbances

The GAST model referred to in the Neptune orbiter attitude control system chapter was updated to the relevant modelling parameters of Uranus.

Disturbances were similar to those around Neptune. The magnetic disturbances were slightly higher due to the closer distance to the dipole centre at periapsis passes and the stronger dipole of Uranus.

Example of disturbances torques during periapsis passes is provided in the plots below:

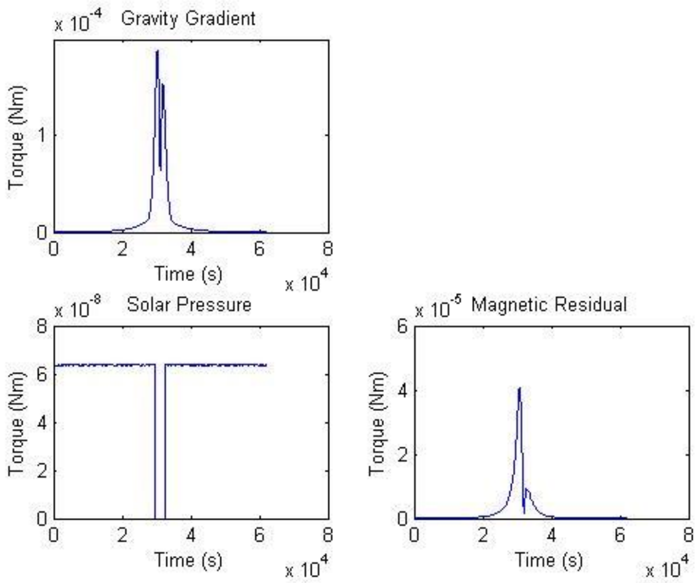

Figure 26-1: Low-periapsis orbit, periapsis pass, inertial pointing. Disturbance torque contributors

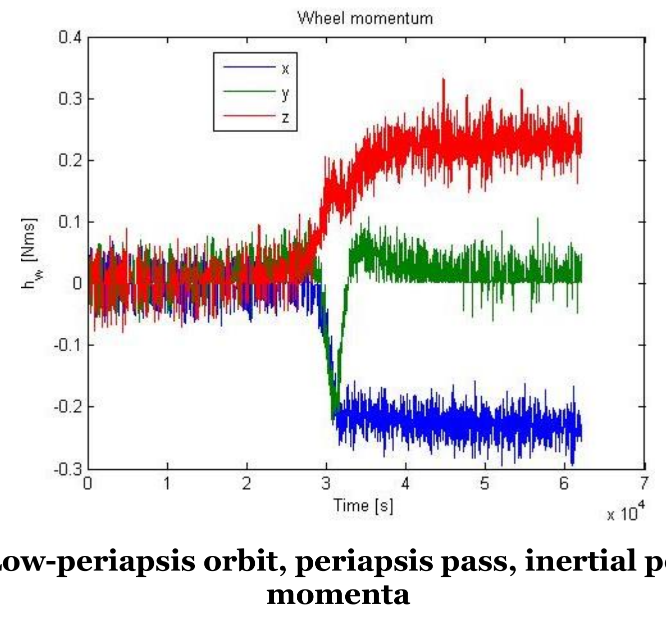

Figure 26-2: Low-periapsis orbit, periapsis pass, inertial pointing. Wheel momenta

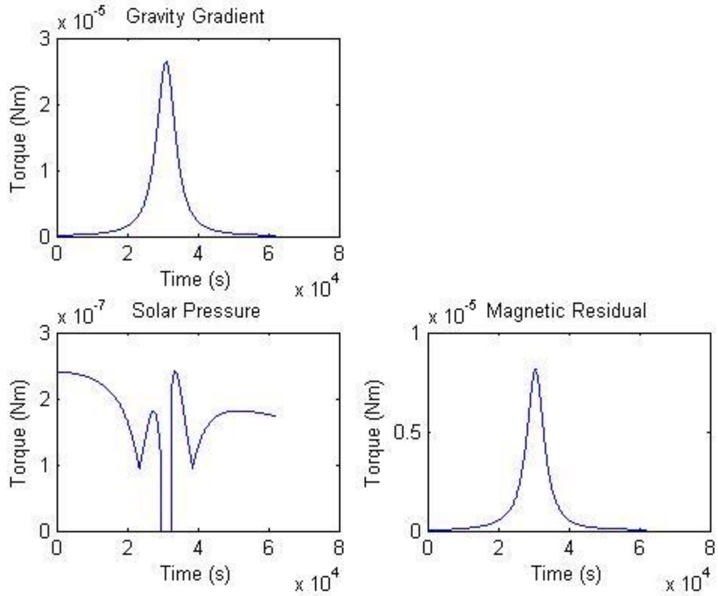

Figure 26-3: Science orbit, periapsis pass, Uranus pointing. Disturbance torque contributors

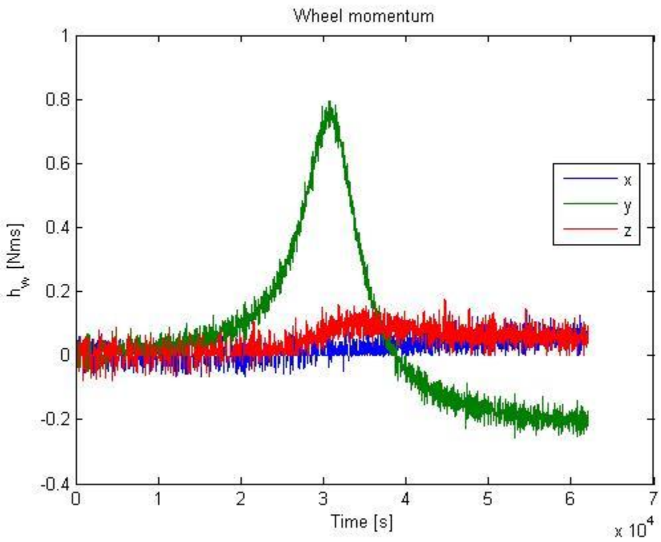

Figure 26-4: Science orbit, periapsis pass, Uranus pointing. Wheel momenta

	Uranus Pointing			Earth Pointing			Earth Pointing (with low pericenter)		
	Pericenter	*Apocenter*	*Orbital mean*	*Pericenter*	*Apocenter*	*Orbital mean*	*Pericenter*	*Apocenter*	*Orbital mean*
Torques (Nm)	Max from simulation over 1 Uranus rotation period		Integrated using AOGNC workbook	Max from simulation over 1 Uranus rotation period		Integrated using AOGNC workbook	Max from simulation over 1 Uranus rotation period		Integrated using AOGNC workbook
Magnetic	1.E-05	6.E-12	1.E-06	1.E-05	6.E-12	1.E-06	4.E-05	6.E-12	5.E-06
Gravity Gradien	3.E-05	1.E-11	2.E-06	5.E-05	2.E-11	6.E-06	2.E-04	2.E-11	3.E-05
Solar Pressure	2.E-07	8.E-08	6.E-08	1.E-07	1.E-07	6.E-08	6.E-08	1.E-07	6.E-08
Aerodynamic	negligible	negligible	negligible	negligible	negligible	negligible	negligible	negligible	negligible
Total Momentum Accumulation (Nms / Uranus-day)	0.2	small	0.2	0.1	small	0.4	0.3	small	2.2
Momentum Max Transient (Nms)	0.8	-> due to rate guidance		0.1	-> due to gravity gradient		0.2	-> due to gravity gradient	

Table 26-1: Environmental torque disturbances summary

As for the Neptune case, the orbital mean torque for gravity gradient seems to be over estimated because the workbook assumes a constant offset from nadir, rather than time varying, and because it does not account for the fact that the high gravity gradient during the half-day prior to periapsis pass is mostly balanced by the gravity gradient the half-day after periapsis pass; this period of time dominates the orbital mean. Therefore, for sizing momentum devices the simulation values for pericenter pass will be used rather than the orbital mean values quoted from the AOGNC workbook.

26.1.3.2.2 Reaction wheel sizing

The Neptune baseline 1 Nms wheels will also suffice for the Uranus mission based on the transient momenta at periapsis pass assuming momentum dumping no more often than once per day.

Due to the high radiation dosage expected, the baseline Honeywell HR04 reaction wheels will almost certainly need parts upgrade and additional global/local shielding. It is possible that less design modifications are required if Rockwell Collins Deutschland wheels are used, but there is limited public information on their wheels in this class.

26.1.3.2.3 Thruster sizing

As per Neptune mission.

26.1.3.3 Attitude control propellant

The input assumptions for the attitude control propellant budget are the same as for Neptune with the exception of the total mission Delta-V mass, which is significantly smaller:

- 2135 m/s total main engine usage (vs 2618 m/s for Neptune mission).

With the Uranus orbits/disturbances and the delta-V mentioned above, the following attitude control propellant is required:

All values in kg	Mass	w/ 100% margin
Attitude control during main engine firings	10.2	20.4
Control overshoot margin	1.02	2.04
Slews transfer phase	0.05	0.1
Slews science phase	1.7	3.4
Planet/moon fly-by rate assist, transfer phase	0.2	0.4
Moon fly-by rate assist pointing, science phase	0.05	0.1
Spin-stabilize/recovery	1.3	2.6
3-axis stabilized safe mode	0.05	0.1
Wheel-momentum dump Neptune-nadir attitude	0.1	0.2
Wheel-momentum dump communications or moon-pointing attitude	0.1	0.2
TOTAL		29.5

Table 26-2: Attitude control propellant budget

Note that the propulsion subsystem has allocated much more than this (roughly double) for attitude control propellant to remain conservative and to help offset the optimistic assumption of steady state specific impulse used to compute the above budget.

26.1.4 List of Equipment

Unchanged from Neptune design.

26.1.5 Options

See Neptune design.

Additional option for Uranus mission:

- Replacement of coarse rate sensor with an extra Airbus Astrix 1090 for safe mode only.

26.1.6 Technology Needs

Technology Needs						
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
X	RW_HR04	Reaction wheels (1 Nms)	Honeywell (US) or	7		Will require parts upgrade to improve lifetime (5 years) and radiation hardness (20 krad is listed)
	RW	Reaction wheels (1 Nms)	Rockwell Collins Deutschland (Germany)	7		Limited information on mass/power, lifetime, radiation, etc. for their 1 Nms wheels. Not clear if product line active and whether wheels can meet lifetime or radiation requirements of Ice Giants. Possible tech development to meet Ice Giant needs.
X	GYRO_Sireus	Safe mode Coarse rate sensor	TAS (UK)	7		Possible parts upgrade required and/or local shielding increase to meet radiation requirements.

27 URANUS GNC

27.1 Atmospheric Probe

The Probe is identical for the Neptune and Uranus missions. Please refer to the Neptune GNC Chapter.

28 URANUS POWER

28.1 Atmospheric Probe

All aspects are unchanged from the Neptune case.

28.2 Orbiter

All aspects are unchanged from the Neptune case with the following points to be noted:

Shorter transfer time

The approx. 2 years shorter transfer time to Uranus vs. Neptune will result in a slightly greater power output from the RTGs at the time of arrival at the target planet. According to the predicted power degradation characteristics of the eMMRTG, this additional power will be approximately 6 W per RTG, 18 W in total.

However, because

- the science mission would likely be extended in this case, making the total mission duration the same in both cases (15 years) and
- 6 W is less than the current uncertainty on the EODL power of the eMMRTG,

it is decided to leave the power budget and sizing results unchanged w.r.t. the Neptune case.

Greater radiation dose (TID)

Increase TID for the Uranus mission may lead to the requirement for extra shielding and/or specific qualification for the PCDU and the cell balancing electronics of the batteries.

29 URANUS TELECOMMUNICATIONS

29.1 Atmospheric Probe

For the probe, requirements and design drivers, assumptions, trade-offs, and baseline design are the same as for the Neptune Probe.

The main difference is the maximum slant range for sizing the Probe-to-Orbiter link. In this respect, the bitrate as function of the distance has been computed as shown in Figure 29-1. According to mission analysis, the worst case distance is about 25,000 km, hence a bit rate of 4.65 kbps is achievable.

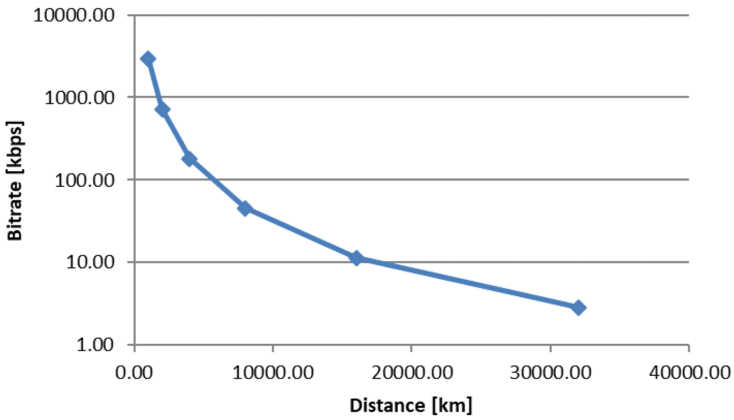

Figure 29-1: achievable bitrate as function of the distance

29.2 Orbiter

For the Orbiter, the requirements and design drivers, assumptions, trade-offs, and baseline design are the same as for the Neptune Orbiter. In particular, it has been considered that a common procurement of the two orbiters' communication subsystem could be done, and thus the Neptune design can be adapted to Uranus by using the same RF output power, and thus increasing the link budget margin.

30 URANUS DATA HANDLING

30.1 Atmospheric Probe

The Probe for Uranus is identical to the probe for Neptune

30.2 Orbiter

The requirements for the Orbiter are the same as for the Neptune Orbiter with the exception of the potential for radiation shielding.

	Data Handling Sub-System Requirements
Req. ID	**Statement**
DHS-0100	The orbiter DHS shall cope with the worst case radiations conditions related to Uranus for which sensitive units shall be provided with 10mm of Al shielding

In all other respects the design of the Orbiter for the Uranus mission is the same as for the Neptune mission.

31 URANUS THERMAL

31.1 Atmospheric Probe

A dedicated thermal analysis of the Uranus probe has not been addressed within this study. Based on the outcome of the PEP study in 2010, entry conditions may be fairly similar to the conditions assumed for Neptune. Due to the comparable atmosphere and potentially comparable entry velocities, the Neptune design might eventually be re-used to some extent. However, a dedicated study will be required to confirm that. Certainly the restrictions in terms of material and facility limitations will be same.

31.2 Orbiter

For the Uranus orbiter the same requirements and assumptions as for Neptune are considered applicable. In particular, no changes in the power dissipation and no changes of the outer surface area of the orbiter are assumed. The Uranus orbiter design is therefore considered to be the same than for Neptune.

32 URANUS AEROTHERMODYNAMICS

The Aerothermodynamics for the Uranus Probe are considered to be identical to the Neptune Probe for the purposes of this study.

33 URANUS EDS PARACHUTE

The Uranus EDS Parachute is designed identical to the Neptune EDS Parachute. Please refer to the Neptune chapter for details.

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34 TRITON LANDER

34.1 Triton Facts & Figures

Triton is orbiting Neptune on an orbit that is circular at a semi-major axis of 355,000 km (somewhat less than our Moon). It has a period of 5.88 days (our Moon: 27.3 days), which is equal to the body rotation period (bound rotation, as is the case for our Moon).

The same face always faces Neptune and day and night each last almost 3 days. The orbit is retrograde with an inclination of ~120 deg with respect to the Ecliptic and ~157 deg with respect to Neptune equator (the orbit of our Moon is prograde).

If the orbit of a Neptune orbiter is optimised for Triton observations, it will also have to be retrograde.

Triton is the seventh largest planetary satellite in the solar system, after the four Galilean moons around Jupiter, Saturn's Titan and our Moon.

- Its diameter is 2706.8 km, its radius 1353.4 km (our Moon: 1738 km)
- Its density is ~2 g/cm^3, indicating that a significant portion of the body is water ice
- Its mass is 29% of that of our moon, so the specific gravitational parameter is ~1429 km^2/s^2
- Velocity on a low, circular orbit is around ~1 km/s (our Moon ~1.65 km/s), so **landing is non-trivial**
- **This puts a hard lower limit on the delta-v required for soft landing**
- Escape velocity on surface: 1.455 km/s.

34.2 Requirements and Design Drivers

The following main design drivers have to be considered for a Lander mission at Triton:

- Release strategy: from orbit around Triton or during flyby only. This has a strong impact on the delta V
- Low atmospheric density, implying that a propulsion-only descent and landing is assumed
- Need for throttled / pulsed propulsion capabilities in a closed-loop GNC system for the final descent manoeuvre (technology gap)
- Possible need of reconnaissance imaginary created by another mission to enable high level selection of safety areas[1]
- Instruments/science (during descent and surface operations): Mass / power / data / temperature/ Operations timeline
- Available communications window(s) duration.

[1] Europa Lander is assuming that they will get images from Clipper beforehand so after the main braking burn it is possible to know where they are with respect to the planet surface.

	Main Requirements	
Req. ID	**Statement**	**Parent ID**
MI-010	The Triton Lander shall land a payload of 11.18 kg	
MI-020	The Triton Lander shall be released from Triton fly-by	
MI-030	The Triton Lander shall perform a soft landing manoeuvre of 4637 m/s	
MI-040	The Triton Lander shall operate during one week of lifetime	

34.3 Assumptions

During the Triton Lander Assessment, the following assumptions were made:

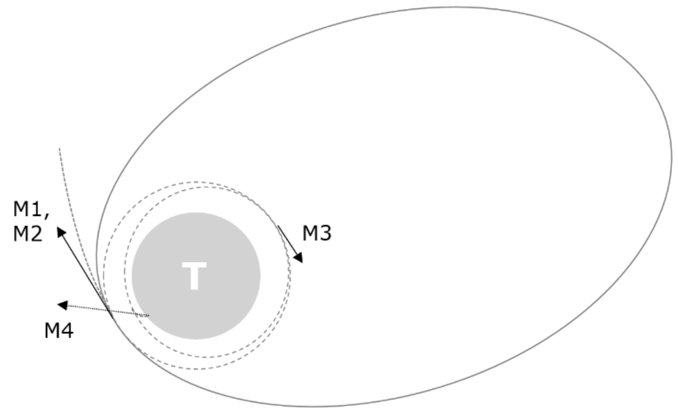
Assumptions	
1	Release from Triton Flyby is considered as baseline strategy, as this imposes less constraints on the orbiter side
2	There is no atmospheric contribution to braking (or heating)
3	It is assumed that the engine and the corresponding propulsion system including the GNC control, throttled or pulsed, required for the final part of the landing is available (technology development)
4	Instrument mass is the bigger of the 2 envelopes defined by the Study Science Team (11.18 kg)
5	Science measurements are taken during 1 week of operational lifetime on the surface of Triton
6	It is assumed that the communication window is such that the communication design for the probe case is valid. A reverse analysis is performed in order to identify the constraints that this implies (in terms of link geometry)
	The dry mass of the system used for the calculation of the propellant mass is 473 kg.

34.4 Trade-Offs

Two Mission analysis scenarios have been considered, addressing two lander release options:

- Release during Triton flyby
- Release from orbit around Triton (50x50km)

34.4.1 Release during Triton Flyby



Direct separation from Triton hyperbolic approach (during flyby) at 3.6 km/s

- 4 manoeuvres:
    - M1 = 2582 m/s
    - M2 = 282 m/s
    - M3 = 7 m/s
    - M4 = 1150-1250 m/s
- For M1 assume 10 to 20% gravity losses
- **Total worst-case delta-v= 4637 m/s**
- **Available system dry mass of ca. 60 to 68 kg* (after system margin)**

* this includes all system elements, including descent propulsion system, GNC, landing airbags, payload, etc

(Isp=300s)

Figure 34-1: Lander release during Triton flyby

- Lander has very significant delta-v capability
- Landing sequence with several burns using the lander propulsion system:
    - Manoeuvre 1: Capture into eccentric Triton orbit. 50x5000 km obit proposed. Orbit is low enough to ensure insertion despite manoeuvre execution errors
    - Manoeuvre 2: Circularisation to 50x50 km orbit to ensure precise targeting to envisaged landing site
    - Manoeuvre 3: Deorbit manoeuvre, lowers periapsis in preparation for landing
    - Manoeuvre 4: Soft landing on surface allowing closed-loop guidance, hazard avoidance etc.
- The chosen strategy will:
    - Allow staging, if required
    - Permit landing at any location on the surface, independently of the hyperbolic arrival conditions
    - Eliminate manoeuvre execution errors and maximise probability of success
    - Simplify GNC by ensuring that most of the delta-v is NOT part of the critical landing burn
- The chosen strategy will not:
    - Reduce the large amount delta-v required.

Manoeuvre 1 is the only manoeuvre in the sequence that depends on the design of the Neptune orbiter tour:

- For v-inf of 3.6 km/s: Man. 1 size: 2582 m/s
- For v-inf of 4.6 km/s: Man. 1 size: 3526 m/s
- For v-inf of 2.6 km/s: Man. 1 size: 1675 m/s

- Depending on the thrust/mass ratio, gravity losses of 10-20% are to be expected.

Flyby No.	Body	TOF (days)	Date (ET)	S/C Rev	Body Rev	Orbit Period (days)	Flyby Alt (km)	LST	Lat. (deg)	W Lon. (deg)	V-inf (km/s)	V (speed) (km/s)
1	Triton	35.3	2044-02-01 22:21:22.40	1	6	35.26	100	22:41:25	-19.1	268.4	3.60	3.86
2	Triton	17.6	2044-02-19 13:29:21.50	1	3	17.63	100	23:50:47	-35.9	285.6	3.60	3.86
3	Triton	11.8	2044-03-02 07:34:40.90	1	2	11.75	100	00:49:45	-39.7	300.3	3.60	3.86
4	Triton	17.6	2044-03-19 22:42:40.00	1	3	17.63	100	08:48:57	-43.5	60.0	3.60	3.86
5	Triton	14.7	2044-04-03 15:19:18.16	1.1	2.5	13.11	1667	23:58:45	-14.8	287.4	3.60	3.73
6	Triton	11.8	2044-04-15 09:24:37.55	1	2	11.75	100	05:41:59	45.5	193.2	3.60	3.86
7	Triton	23.5	2044-05-08 21:35:16.34	3	4	7.84	100	08:35:00	10.0	236.4	3.60	3.86
8	Triton	5.9	2044-05-14 18:37:56.03	1	1	5.88	100	07:57:01	7.8	226.8	3.60	3.86
9	Triton	5.9	2044-05-20 15:40:35.73	1	1	5.88	100	16:39:18	-39.8	357.3	3.60	3.86
10	Triton	5.9	2044-05-26 12:43:15.43	1	1	5.88	100	04:39:13	39.8	177.3	3.60	3.86
11	Triton	14.7	2044-06-10 05:19:55.76	1.8	2.5	8.27	199	21:59:00	18.1	77.2	3.60	3.85
12	Triton	11.8	2044-06-21 23:25:15.15	1	2	11.75	100	08:25:54	-30.4	53.9	3.60	3.86
13	Triton	11.8	2044-07-03 17:30:34.53	1	2	11.75	100	04:39:50	-25.3	357.3	3.60	3.86
14	Triton	11.8	2044-07-15 11:35:53.92	1	2	11.75	100	16:39:41	25.3	177.2	3.60	3.86
15	Triton	17.6	2044-08-02 02:43:53.00	1	3	17.63	100	13:47:17	19.3	134.1	3.60	3.86
16	Triton	17.6	2044-08-19 17:51:52.07	1	3	17.63	100	16:40:00	39.7	177.2	3.60	3.86
17	Triton	11.8	2044-08-31 11:57:11.46	1	2	11.75	100	20:20:09	38.0	232.2	3.60	3.86
18	Triton	17.6	2044-09-18 03:05:10.53	2	3	8.82	100	20:28:14	46.6	234.1	3.60	3.86
19	Triton	23.5	2044-10-11 15:15:49.28	3	4	7.84	100	18:56:34	61.2	211.1	3.60	3.86
20	Triton	11.8	2044-10-23 09:21:08.66	1	2	11.75	100	12:20:11	21.4	112.0	3.60	3.86
21	Triton	20.6	2044-11-12 23:00:26.47	1.1	3.5	19.09	217	10:05:03	-35.0	78.1	3.60	3.85
22	Triton	11.8	2044-11-24 17:05:45.84	1	2	11.75	100	11:30:26	-39.7	279.4	3.60	3.86
23	Triton	17.6	2044-12-12 08:13:44.90	2	3	8.82	100	11:24:40	-54.2	277.9	3.60	3.86
24	Triton	17.6	2044-12-29 23:21:43.96	2	3	8.82	100	16:43:40	-80.6	357.6	3.60	3.86
25	Triton	23.5	2045-01-22 11:32:22.69	3	4	7.84	100	09:52:07	-74.0	254.6	3.60	3.86
26	Triton	5.9	2045-01-28 08:35:02.38	1	1	5.88	100	09:20:11	36.2	246.6	3.60	3.86
27	Triton	5.9	2045-02-03 05:37:42.06	1	1	5.88	100	04:44:05	83.4	177.5	3.60	3.86
28	Triton	23.5	2045-02-26 17:48:20.79	3	4	7.84	100	20:58:03	-32.6	61.0	3.60	3.86
29	Triton	11.8	2045-03-10 11:53:40.15	1	2	11.75	100	22:05:05	30.9	77.7	3.60	3.86
30	Triton	17.6	2045-03-28 03:01:39.20	2	3	8.82	100	08:48:41	48.9	238.5	3.60	3.86
31	Triton	23.5	2045-04-20 15:12:17.92	3	4	7.84	100	06:56:02	60.3	210.3	3.60	3.86
32	Triton	17.6	2045-05-08 06:20:16.95	2	3	8.82	100	18:56:50	-60.3	30.4	3.60	3.86

Table 34-1: Triton baseline flyby tour, shadowed rows show night time flybys

Manoeuvre 2:

- Circularization to 50x50 km orbit
- Nominal delta-v: 282 m/s
- Gravity losses are low, as manoeuvre can be split into parts

Manoeuvre 3:

- Deorbit to ~10x50 km orbit
- Nominal delta-v: 7 m/s

Manoeuvre 4:

- Soft landing
- Delta-v around 1150-1250 m/s depending on thrust/mass ratio, GNC & hazard avoidance strategy.

Total budget (Example for v-inf 3.6 km/s)

- Low: 2582*1.1 + 282 + 7 + 1150 m/s = 4279 m/s
- High: 2582*1.2 + 282 + 7 + 1250 m/s = 4637 m/s
- The delta-v requirements on EDL is a Key driver for the mission.

34.4.2 Release from orbit around Triton (50x50km)

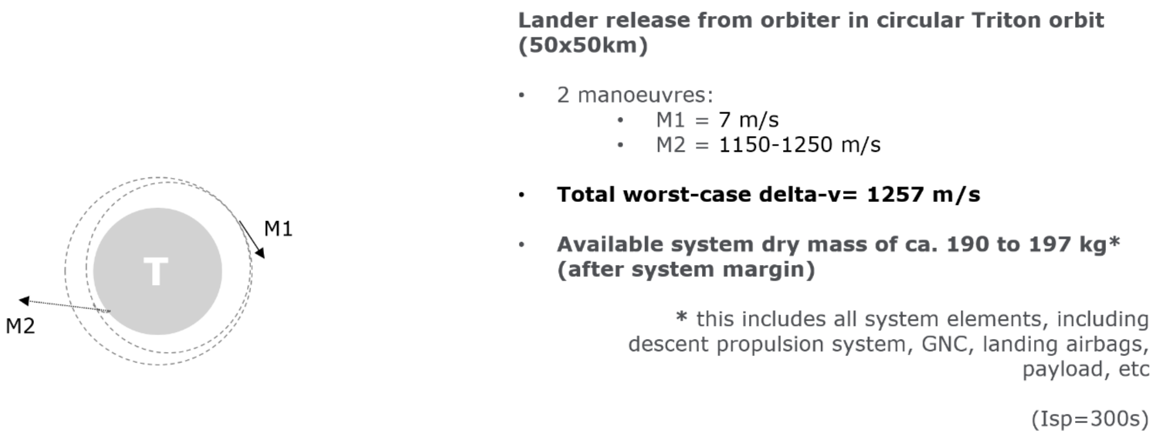


Figure 34-2: Lander release from Triton orbit

Assuming that an orbiter has placed the lander in low Triton orbit, **and not considering the impact on the orbiter manoeuvre into Triton orbit,** the delta-v for landing is reduced to manoeuvres 3+4:

Manoeuvre 3:

- Deorbit to ~10x50 km orbit
- Nominal delta-v: 7 m/s

Manoeuvre 4:

- Soft landing
- Delta-v around 1150-1250 m/s depending on thrust/mass ratio, GNC & hazard avoidance strategy.

Manoeuvre 4 is different from all others in the sequence as the main engine must be controlled in closed loop by a GNC system that:

- Measures the imparted and computes the remaining delta-v
- Computes the required thrust direction as function of time into manoeuvre and controls the attitude accordingly
- Regulates the thrust level in the final part close to the surface*
  ** This requires an engine that can be either throttled or pulsed. The requirement for throttled or pulsed operations is specific to landing and does not apply to any of the other phases. Initial ~90% of manoeuvre is executed at full thrust.*
- Measures vertical and lateral velocity and achieves ~0 velocity in all components at ~0 altitude (size of "~" TBD)
- Performs hazard detection and commands avoidance manoeuvres (if applicable)
- Can allow limited pre-touch-down hovering (optional).

Note: Triton gravity at surface: 0.78 m/s² = 1/12.6 g. (our Moon: 1.623 m/s²=1/6 g) and the required MINIMUM thrust level per 100 kg of landed mass is 78 N.

34.5 Baseline Design

The Model Payload for the Triton Lander was put together in order to size the lander resources and to address the payload accommodation. These instruments are a representation of a possible future payload but are not meant to be understood as a pre-selection of instruments for a potential future mission.

Two options for a scientific Model Payload for a Triton Lander were established: the baseline Model Payload (see Table 34-2), and a lighter Model Payload (see Table 34-3).

Table 34-2 lists the baseline instruments, together with their mass (incl. 20% maturity margin), average power consumption, data rate, physical size and their heritage from previous instruments and missions.

Instrument	Mass [kg]	Power [W]	Data rate	Volume envelope [mm]	Notes/Heritage
Imaging system	3.40	12.1	1.2-24 Mbits/s	70x56x36 (single) 112×19×90 (stereo) 70x50x94 80x50x120	Panoramic, microscopic imaging and analysis of the sample composition; three units comprising the system. Based on CIVA/Philae (ROSETTA)
In situ science package	2.35	2.2	180kb/h	TBD	Mechanical, electrical and thermal properties of surface, soil and subsoil. Based on SSP/Huygens MUPUS, SESAME/Philae
Gas Analyser / Mass spectrometer	4.50	14.0	2kb/s	TBD	TOF measurement for Triton's atmosphere; based on JUICE/NIM (part of PEP on JUICE), COSAC, PTOLEMY/Philae
Magnetometer and plasma monitor	0.93	0.9	0.03-4.4kb/s	TBD	Heritage from ROMAP/Philae incl. ions & electrons detectors

Table 34-2: Triton Lander draft Model Payload Definition – 1

The total mass of the baseline Model Payload amounts to 11.18kg, including 20% maturity margin with an average power consumption of ~29.20W (no margin included). This mass presents a maximum value for these instruments, as miniaturization of these instruments has been taking place in recent years. Also, the technological progress since the conception of the e.g. Rosetta instruments would allow for lighter instrument for the

34.5.1 Mission to Triton

More details on some of the instruments are listed here below:

- **Imaging System:** The mass of 3.4 kg is an upper limit as the CIVA instrument on Philae is sharing resources with ROLIS-Philae (a nadir-pointing camera).
- **In situ science package:** The values of the MUPUS instrument on the Rosetta Philae lander were taken as reference. The mass of 2.35 kg also includes the harness. The data rate of 180kb/h is an average for long-term operations with measurements every 20 seconds.
- **Gas Analyser/Mass spectrometer:** The values of the PTOLOMY instrument on the Rosetta Philae lander were taken as reference. The 14W average power consumption is based on 4W used in quiescent phases and 28W as peak power.

Table 34-3 lists the lighter Model Payload, together with their mass (incl. 20% maturity margin), average power consumption, data rate, physical size and their heritage from previous instruments and missions.

Instrument	Mass [kg]	Power [W]	Data rate	Volume envelope [mm]	Notes/Heritage
Imaging system	0.49	4.00	2.1 Mbyte/ image	113×136×81	Based on MASCam/MASCOT (heritage from Philae ROLIS)
In situ science package	0.38	0.31	0.028 kb/s	54×48×26 + 152×152×5 + 109×94×14	Mechanical, electrical and thermal properties of surface, soil and subsoil. Based on SSP/Huygens MUPUS, SESAME/Philae (reduced version).
VNIR spectrometer	1.05	22.00 *	23.3 kb/spectrum	TBD	VNIR miniaturized spectrometer; heritage from MaMISS/ExoMars2020, CIVA-M/Philae (partially)
Magnetometer and plasma monitor	0.32	0.50	0.07 kb/s	TBD	Heritage from ROMAP/Philae incl. ions & electrons detectors

Table 34-3: Triton Lander draft Model Payload - 2

The total mass of this Model Payload amounts to 2.24kg, including 20% maturity margin with an average power consumption of ~26.81W (no margin included).

More details on some of the instruments are listed here below:

- **Imaging System**: Based on MASCam, the camera of the European MASCOT asteroid lander on board Hayabusa 2 (with heritage from Philae ROLIS descent camera). Spectral range 400 nm–870 nm, 4 LEDs (RGB, IR), 1.4–1.8 W for imaging and 4,2–6.4 W for imaging + LED. Alternatively, CIVA/Philae but only partially e.g. the stereo or 2 micron camera heads with filters.
- **In situ science package**: The values are as per MARA/MASCOT, a radiometer (heritage MUPUS-TM/Philae) taken as placeholder. This could also be a reduced version of MUPUS with a passive penetrometer (e.g. Huygens SSP ACC-E) plus sensors.

- **VNIR spectrometer**: Derived from MaMISS on the ExoMars2020 Rover. MaMISS is located inside the drill for borehole science; spectral range 0.4–2.2 µm; 22W is peak for science mode; average power is TBD.

The Lander design has been initiated with a top-down assessment, exploring potential reference missions in order to derive rough scaling factors in order to get a feeling for the mass range of the system.

	Ice Giants // Triton	Huygens // Titan
Surface gravity (m/s^2)	0.779	1.352
Surface pressure (mbar)	0.014	1500
Approach velocity (km/s)	3.6	6
Total probe mass (kg)	350	318.3
Payload mass (kg)	11.18	47.6

Table 34-4: Rough scaling factor with Huygens Titan lander

Assuming a **total lander system wet mass** of **350 kg** (same ball-park number of the probe design in the frame of the CDF Study), a top down estimate of the payload landed mass was derived, taking a reference case of a known lander mission.

Using the same scaling ratios, the required Lander wet mass to land the instrument package defined by the science team was computed.

The scaling exercise, performed for Lander release from Triton flyby, indicated the following figures:

	Reference case	Scaled Triton lander (top-down)	Scaled Triton lander (bottom-up)
Lander wet mass (kg)	474	350	2680.4
Propellant mass (kg)	350	277	2125.9
Lander dry mass* (kg)	124	72	554.5
- Descent module	*109*	*63.3*	*487.4*
- Lander station	*15*	*8.7*	*67.1*

** Still includes system margin*

	Reference case	Scaled Triton lander (top-down)	Scaled Triton lander (bottom-up)
Lander station (kg)	15	8.7	67.1
- Airbags	*5.5*	*3.2*	*24.6*
- Lander platform	*7*	*4.1*	*31.3*
- Payload	*2.5*	*1.5*	*11.18*

Table 34-5: Scaling exercise for Lander release from Triton flyby

The scaling exercise, performed for Lander release from 50x50 km Triton orbit, indicated the following figures:

	Reference case	Scaled Triton lander (top-down)	Scaled Triton lander (bottom-up)
Lander wet mass (kg)	474	350	850.0
Propellant mass (kg)	350	122	295.5
Lander dry mass* (kg)	124	228	554.5
- Descent module	*109*	*200.0*	*487.4*
- Lander station	*15*	*28.0*	*67.1*

** Still includes system margin*

	Reference case	Scaled Triton lander (top-down)	Scaled Triton lander (bottom-up)
Lander station (kg)	15	28	67.1
- Airbags	*5.5*	*10.3*	*24.6*
- Lander platform	*7*	*13*	*31.3*
- Payload	*2.5*	*4.7*	*11.18*

Table 34-6: Scaling exercise for Lander release from 50x50 Triton orbit

A very rough scaling function was derived, for both mission analysis scenarios, and it is shown hereafter:

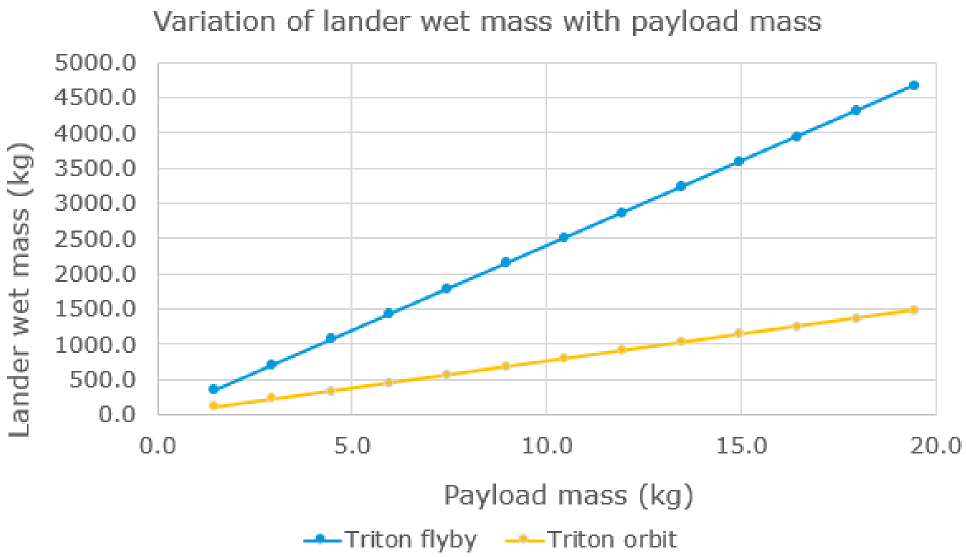


Figure 34-3: Rough scaling function for both mission analysis scenarios

The scaling factor is based on a single point of reference, and does not consider the scaling function rules for the propulsion subsystem hardware with the propellant mass. Also, it is assumed that airbags scale linearly which is far from being real, therefore there is no intention of correctness with providing these figure, but rather to give quick ROM rules of thumb.

A bottom up mass budget has been built, based on a preliminary estimate at subsystem level for GNC, Propulsion, Mechanisms (landing gear) and Power. For the structures, a 20% of the System Dry Mass is assumed. For Communications and Data Handling the same design of the Neptune probe is retained.

Subsystem	Switch	Probe Mass Budget		Mass [kg]	
GNC	Product	Guidance Navigation and Control		27.00	no redundancy
COM	Product	Communications		12.31	from probe
CPROP	Not used	Chemical Propulsion		127.00	wet mass of 350kg assumed
DH	Product	Data-Handling		1.00	from probe
EPROP	Not used	Electric Propulsion		0.00	
INS	Product	Instruments		11.18	
MEC	Product	Mechanisms		20.00	3% of lander mass, assumed landing systems that also needed to ascent
PWR	Product	Power		55.00	2 weeks mission assumed – RTG Option
STR	Product	Structures		90.00	40kgl anding legs + 30 kg structure + margin
EDL	Product	Entry, Descent and Landing		24.60	airbags
TC	Product	Thermal Control		14.19	from probe
		Harness	5%	19.88	
		Dry Mass w/o System Margin		**402.15**	
		System Margin	20%	80.43	
		Dry Mass incl. System Margin		**482.58**	
		CPROP Fuel Mass		654.13	fly-by option, 4322 km/s Delta V
		CPROP Fuel Margin	0%	0.00	
		CPROP Oxidizer Mass		855.07	
		CPROP Oxidizer Margin	0%	0.00	
		CPROP Pressurant Mass		4.72	
		CPROP Pressurant Margin	0%	0.00	2% margins already included
		EPROP Propellant Mass			
		EPROP Propellant Margin	2%	0.00	
		Total Wet Mass		**1996.50**	

Table 34-7: Mass budget

34.5.2 GNC

34.5.2.1 GNC design drivers

34.5.2.1.1 Soft landing requirements

The main objective of the GNC is to achieve certain performances of velocity and attitude orientation in the instant of touchdown in order to ensure structural compatibility with the landing system (landing legs, crushable structure or airbags) and avoid roll-over.
For this specific mission there is another stringent requirement for low contamination of the landing area with the engine's plumes. In the case of the Europa Lander mission, NASA's way of dealing with this requirement is the use of a large sky-crane landing system (cable of 15 meters) and thrusters canted 30 degrees during sky-crane manoeuvres (RD[55]).

An alternative solution could be to switch-off the engines at 15 meters and design a crushable structure or landing legs capable of sustaining the loads derived from this free-fall. The following table shows a comparison of impact velocities for different drop altitudes and different planets.

	Triton	Europa	Moon	Mars	Earth
Radius [km]	1353.4	1561	1737	3390	6378
Surface gravity [m/s2]	0.779	1.314	1.62	3.711	9.806
Surface pressure [Pa]	1.4-1.9	1.0E-12	0	800	101325
mu [km3/s2]	1427	3202	4888	42647	398914

Table 34-8: Radius, atmospheric pressure and gravitational parameter of different moons and planets

Drop Altitude	Impact Velocities [m/s]					Time to touchdown [s]				
[m]	Triton	Europa	Moon	Mars	Earth	Triton	Europa	Moon	Mars	Earth
1	1.2	1.6	1.8	2.7	7.3	1.6	1.2	1.1	0.7	0.5
2	1.8	2.3	2.5	3.9	8.7	2.3	1.7	1.6	1.0	0.6
3	2.2	2.8	3.1	4.7	9.6	2.8	2.1	1.9	1.3	0.8
10	3.9	5.1	5.7	8.6	13.0	5.1	3.9	3.5	2.3	1.4
15	4.8	6.3	7.0	10.6	14.4	6.2	4.8	4.3	2.8	1.7

Table 34-9: Impact velocities in different planets depending on initial drop altitude

For the case of Schiaparelli Mars landing, the crushable structure was designed to withstand a vertical velocity of 4.8 m/s, equivalent to a free fall from 3 meters altitude. In the case of Triton, a similar vertical velocity is encountered if the free fall starts from 15 meters.
But in contrast with Mars, the free fall from 15 m in Triton lasts 6.2 seconds which imposes a constraint on the initial orientation and residual angular rate at the moment of engines switch off. The attitude with respect to the local vertical at touchdown together with the lateral velocity determines whether or not the roll over will occur.

A further assumption on the geometry of the lander is needed to complete the analysis. The following picture shows a schematic of two possible lander designs: with a crushable structure or with landing legs.

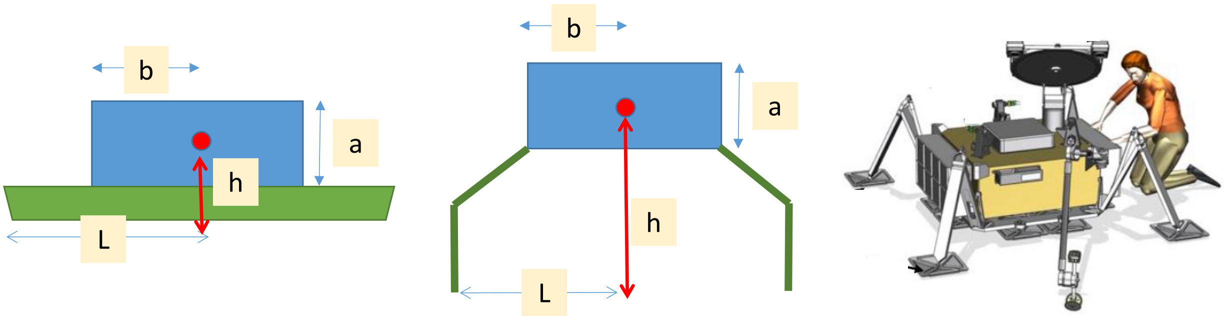


Figure 34-4: Possible lander configurations (left, centre) and Europa Lander NASA design (right)

	Geometric/MCI assumptions	
	Crushable	Landing Legs
a [m]	0.3	0.3
b [m]	0.3	0.3
h [m]	0.2	0.8
L [m]	0.6	0.8
mass [kg]	80	80
Ixx [kg*m2]	10	10

Table 34-10: Assumptions for MCI and lander configurations

A simplified assessment of the dynamic landing stability can be performed based on angular momentum considerations and additional simplifications detailed hereafter. These analyses feature very simple assumptions (no rebounce, no model of the leg dampers, etc.), so conclusions must be considered with care, and more detailed analyses will be needed in future phases.
The first assumption is a two-dimensional movement, as presented in Figure 34-5. Additionally it is assumed that at the moment of impact, the leg or crushable structure will encounter some kind of obstacle which would prevent it from sliding and the whole vehicle will start a rotational movement around the impact point (point A in Figure 34-5).

The initial angular rate $\dot{\theta}$ after touch down will be computed applying the conservation of angular momentum between the two instants prior and after the impact:

$$\dot{\theta} = \dot{\theta}_{SOT} + \frac{m \cdot v_h \cdot \sqrt{l^2 + h_{CoG}^2} \cdot \sin(\theta)}{I_{XX@CoG} + m \cdot (l^2 + h_{CoG}^2)}$$

where $\dot{\theta}_{SOT}$ is the angular rate prior to impact, which is assumed to be the same as the angular rate at the switch-off of the thrusters (SOT), m is the mass of the lander composite, v_h the horizontal velocity prior to impact. The denominator represents the moment of inertia around the impact point A (see Figure 34-5)

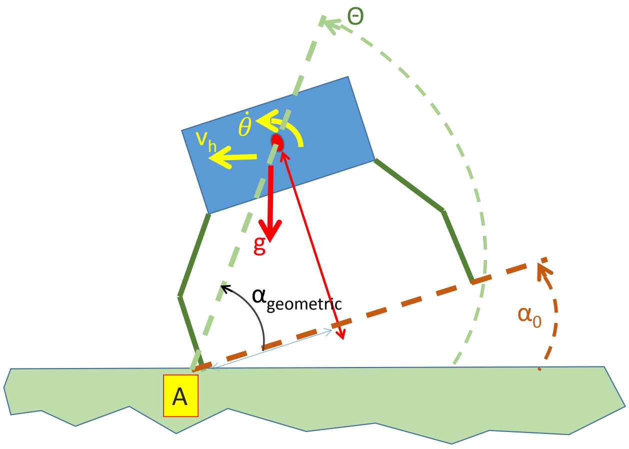


Figure 34-5: Parameter definitions for the preliminary landing stability assessment

During this rotational movement the gravitational acceleration of Triton creates an angular deceleration which tends to counteract the initial angular velocity. This angular acceleration depends on θ and it is computed using the following formula:

$$\ddot{\theta} = -\frac{g \cdot m \cdot \sqrt{l^2 + h_{CoG}^2} \cdot \cos(\theta)}{I_{XX@CoG} + m \cdot (l^2 + h_{CoG}^2)}$$

The above equations have been integrated over time for different initial angular offset with respect to the surface at the moment of touchdown (αTD). Note that the αTD is related to the initial angular offset at SOT (α_SOT) with the following formula:

$$\alpha TD = \alpha\_SOT + (time\_free\_fall) \times (\dot{\theta}_{SOT})$$

Note that in all the analyses it is assumed that the surface is a geodetic surface, meaning that it is perpendicular to the gravitational acceleration. The roll-over condition occurs when the angle θ exceeds 90 degrees.

Figure 34-6 shows the evolution of the angle θ versus time for the crushable structure configuration, assuming an initial angular rate $\dot{\theta}_{SOT}$ of 1 deg/s and two different lateral velocities: 1.5 and 2 m/s. Figure 34-7 shows similar simulations for the landing legs configuration.

Table 34-11 shows in a parametric analysis which is the allowable attitude error at SOT for given lateral velocities and attitude rates at SOT.

Based on previous Moon and Mars studies involving vision based navigation system, the expected performance in terms of lateral velocity knowledge can be in the order of 0.25-0.5 m/s and the final lateral velocity achieved can be controlled down to 1.5 to 2 m/s. It is also assumed that at SOT an attitude error of 1 or 2 degrees and angular rate error residual of 1 deg/s is achievable. This attitude error performance imposes a stringent constraint on the thruster actuation accuracy and its feasibility should be better assessed in future studies. With such assumptions, the crushable structure configuration as defined in Table 34-10 seems to be a feasible option in terms of GNC for a drop altitude of 15 meters above the surface.

On the contrary, if a landing system with legs is deemed necessary for whatever reason, and the required altitude of the CoG with respect to the ground is 0.8 m, this would impose a big challenge on the navigation and control performance, which should be able to reduce the lateral velocity to 0.5 m/s and angular rate of 1 deg/s at SOT.

This analysis will need to be refined once a proper design of the lander is identified, but it can already be used as an indication of the challenges that might be encountered.

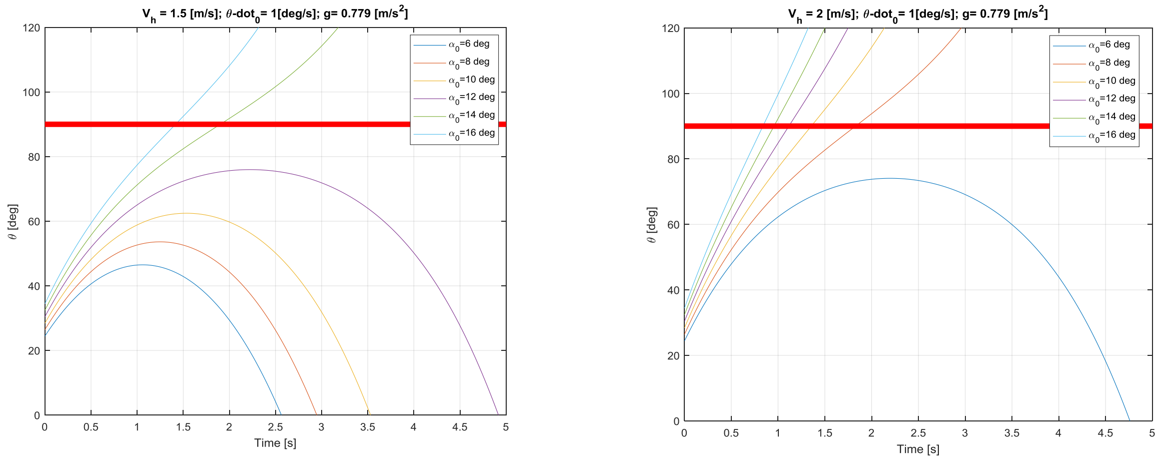


Figure 34-6: Crushable structure configuration. Angular evolution after impact for initial $\dot{\theta}_{SOT}$ = 1 deg/s and v_h = 1.5 m/s (left) and v_h = 2.0 m/s (right)

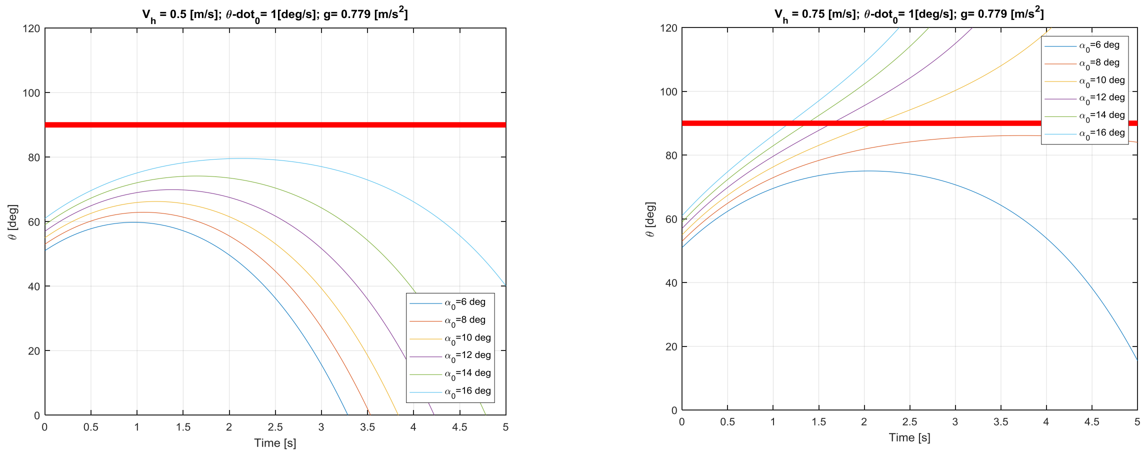


Figure 34-7: Landing leg configuration. Angular evolution after impact for initial $\dot{\theta}_{SOT}$ = 1 deg/s and v_h = 0.5 m/s (left) and v_h = 0.75 m/s (right)

vh [m/s]	h_CoG [m]	L [m]	Mass [kg]	Ixx@CoG [kg*m2]	$\dot{\theta}_0$ [deg/s]	Maximum α_0 [deg]	Maximum α_{0_SW} [deg]
2	0.2	0.6	80	10	1	6	0
2	0.2	0.6	80	30	1	10	4
2	0.2	0.8	100	30	1	9	3
2	0.2	0.8	80	10	1	12	6
2	0.2	0.8	80	10	1.5	12	3
1.5	0.2	0.6	80	10	1	12	6
1.5	0.2	0.6	80	30	1	16	10
1.5	0.2	0.8	100	30	1	15	9
1.5	0.2	0.8	80	10	1	18	12
1.5	0.2	0.8	80	10	2	18	6
0.5	0.8	0.6	80	10	1	8	2
0.75	0.8	0.8	80	30	1	8	2

Table 34-11: Max. allowable attitude error at SOT (αSOT) for a given $\dot{\theta}_{SOT}$ and v_h

34.5.2.2 Thrust level requirements for soft-landing manoeuvre

In both of the mission analysis scenarios considered (delivery from fly-by or from orbit) the last manoeuvre (M4 or M2 respectively) is called "soft landing manoeuvre" and requires a total delta-v in the order of 1150-1250 m/s to slow down the vehicle from the periapsis velocity of a 10x50 km orbit to touchdown conditions. The manoeuvre is planned to be performed in two phases: a first phase where the biggest part of the dv is performed (using a higher thrust level) and a second phase where the last 200 m/s are provided with a lower thrust level, aiming to have a thrust to weight ratio of 1 prior to the switch-off thrusters (SOT) event.

This strategy is similar to the one proposed by Europa Lander (see section 34.7). In the case of the Europa Lander, the arrival to Europa is done via a fly-by approach which can be assumed to be equivalent to an arrival from a 6 x 6500 km high elliptical orbit. For such an orbit, the passage through the periapsis occurs very rapidly and therefore not only the gravity losses will increase dramatically if the T/W is small, but also the whole strategy might become unfeasible if the periapsis braking manoeuvre is stretched over time. As a result, for the Europa Lander the thrust level proposed is 70 kN (i.e non-dimensional T/W = 16.7), the burn lasts approximately 60 seconds and delivers a delta-v of 1750 m/s. Assuming that the manoeuvre is started at periapsis, Table 34-12 shows the condition at the end of the burn in terms of true anomaly, altitude and flight path angle (FPA), being the latter the main contributor for the gravity losses.

Just for comparison, Table 34-12 shows also the initial orbit for the Triton lander and what would be the conditions 60 seconds after the periapsis passage. As it can be seen, the FPA is almost two order of magnitudes smaller, showing that the delta-V losses for the Triton case would be negligible in the manoeuvre was to last only 60 seconds.

Planet	Perig Alt	Apog Alt	Perig Vel	True Anomaly (v)	FPA @ v	Alt @ v	T from perigee to v
	[km]	[km]	[m/s]	[deg]	[deg]	[km]	[s]
Europa	6	6500	1850	2.5	1.01	6.60	60
Triton	10	50	1030	2.5	0.04	10.02	60

Table 34-12: Comparison of initial orbits to start soft landing manoeuvre: Europa Lander vs Triton Lander

Assuming an Isp=300 s, the wet mass of the Triton lander is 111 kg at periapsis, since it has already consumed a considerable amount of propellant to perform the previous manoeuvres. As explained above, the soft-landing 1250 m/s manoeuvre is split in two: 1050 m/s with high T/W thrust (33 kg) and 200 m/s at lower T/W (5 kg). The final mass prior to touchdown is therefore 73 kg. As a result, the minimum required thrust level at touchdown shall be 57 N, to allow a non-dimensional T/W ratio equal to 1.

@First Maneouvre			@Periapsis		@TouchDown				Conditions@end_burn with maneouver starting@periapsis			
Mass	T/W	T/W	Mass	T/W	Mass	T/W	T	(T_Max/T_Min)	Burn time	True anomaly	FPA	Altitude
[kg]	[-]	[N/kg]	[kg]	[-]	[kg]	[-]	[N]	Ratio Request	[s]	[deg]	[deg]	[km]
350	2.5	1.9	111	7.9	73	12.0	680	12.0	142	6.1	0.09	10.11
350	1.7	1.3	111	5.2	73	8.0	**453**	8.0	214	9.2	0.13	10.25
350	1.0	0.8	111	3.3	73	5.0	284	5.0	339	14.5	0.20	10.62
350	0.8	0.6	111	2.4	73	3.6	**205**	3.6	463	19.9	0.28	11.16
350	0.4	0.3	111	1.3	73	2.0	114	2.0	827	35.6	0.48	13.64
350	0.2	0.2	111	0.7	73	1.0	57	1.0	1699	75.0	0.80	24.55

Table 34-13: Triton lander thrust level, throttable requirements and T/W ratios during mission

Table 34-18 shows different possible thrust levels for the Triton lander. For example, if the thrust level selected is 57 N, the non-dimensional T/W at touchdown would be 1, but the same thrust level should be used to perform the manoeuvre at periapsis (1050 m/s), which would last 1699 seconds, equivalent to 75 degrees of true anomaly. The most critical issue is that the non-dimensional T/W during the first manoeuvre would be 0.2, which most likely will provoke gravity losses higher than the 20% currently assumed.

On the contrary, if a single thruster of 453 N would be used, the non-dimensional T/W during the first manoeuvre would be 1.7, which most likely is covered by the 20% gravity losses currently assumed in the delta-v budget (confirmation of this assumption should be done in future steps of the study). The burn duration at periapsis would only last 214 seconds with a final FPA of 0.13 deg, ensuring negligible gravity losses in this manoeuvre. But in order to ensure safe touch down, the engine should be throttleable by a factor of 8.

Another possibility is to use a cluster of engines used in pulse-width-modulation (PWM). This was the solution chosen for Schiaparelli (RD[63]). For a landed mass of approximately 270 kg, Schiaparelli used 3 clusters of 3 CHT-400 hydrazine engines, canted 20 deg. The maximum effective thrust using all the engines in continued mode was 3420 N, while for the final part of the flight, the PWM managed to reduce the total thrust to 1000 N. This is equivalent to a throttleable capability of a factor of 3.4.

For the Triton lander, a Schiaparelli-like strategy would imply a total thrust level of 204 N to be distributed in 3 clusters of either 3 thrusters 22N each. Another option would be to use the Phoenix/Insight configuration which used also 3 clusters but with only 4 engines per cluster, which would result in a total of 12 thrusters 17N. Another option would be 3 clusters with 2 thrusters 35N each.

This option would yield to a 463 s periapsis burn duration (negligible gravity losses) and a non-dimensional T/W of 0.8 at the beginning of the flight. It needs to be further assessed whether this T/W would be covered by the 20% gravity losses margin already included in the budget. If this is the case, this option should be further analysed in future studies to estimate the mass of the propulsion system.

34.5.2.3 Baseline GNC design

The overall operation of the GNC functions is very similar to the one proposed by NASA for the Europa Lander (see Figure 34-14). This is also similar to the PILOT (Precise and Intelligent Landing Using Onboard Technologies) navigation system, which is to be flown on the robotic Luna-Resource (2023) lander mission as part of the ESA cooperation with Roscosmos on exploration. PILOT provides relative & absolute vision based navigation and hazard-detection-and-avoidance (HDA) capabilities by means of:

- Camera Optical Unit (70deg FoV full cone, with a baffle with an exclusion angle of 110deg full cone). Estimated mass @ PDR (maturity + 10% system) ~ 0.8 kg
- LIDAR: scanning LIDAR developed by Neptec, maximum range of 1500m, 40 deg FoV max and scan time of 5 s. Estimated mass @ PDR (maturity + 10% system) ~ 7.0 kg
- Landing Processing Unit (LPU): includes a processor board, and FPGA board, mass memory unit and power distribution unit to power the camera, not the LIDAR. Estimated mass @ PDR (maturity + 10% system margin) ~ 9.0 kg

The navigation camera is used to create hazard maps in terms of texture and illumination, while the LIDAR identifies hazards related to slopes.

The navigation camera will also be used to provide estimation of the relative velocity with respect to the ground by identifying and tracking features of the terrain.

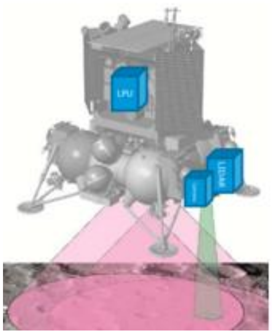
A priori images of Triton might be needed to increase the absolute navigation knowledge with respect to the surface.

In addition an altimeter is needed to provide relative altitude with respect to the ground, since the position estimation purely based on inertial measurement units would not yield the sufficient precision.

The mass and power estimation of such an altimeter has been done using the results obtained from an ESA TRP - Planetary Altimeter breadboard (ABPA) - aiming to analyse, design and bread-board two versions (radar and laser) of a compact, low-power planetary altimeter for use in ESA exploration programmes (Sep-2018).

- RADAR altimeter: Estimated mass (including antennas) ~ 1.7 kg. Power consumption 5.6 W. Operational envelope: 10 m – 2.2 km.
- LIDAR altimeter: Estimated mass (unit box only) ~ 0.6 kg. Power consumption 8.6 W. Operational envelope (max distance limited by test campaign): 18 – 810 m

As a result, the estimated mass budget for the GNC equipment is provided in Figure 34-12. It should be highlighted that no redundancy is included in this equipment list, while the NASA Europa Lander assumes full redundancy of the GNC equipment. In addition, the propulsion system (either main engine with reaction control thrusters or cluster of engines) is not included this equipment list.



	Quantity	Mass per unit, excl margin [kg]	Maturity Level / mission	Margin [%]	Total Mass Inc margin [kg]
Star Tracker	1	2.7	COTS (Sodern Hydra)	5%	2.8
Radar Altimeter	1	1.7	Planetary Altimeter TRP	20%	2.0
European IMU	1	5	ExoMars 2020	5%	5.3
LIDAR	1	5.8	PILOT	20%	7.0
Camera	1	0.7	PILOT	20%	0.8
Landing Processor Unit	1	7.5	PILOT	20%	9.0
	Total Mass with margin [kg]				27

Figure 34-8: PILOT inside Lunar Resource (left) and Triton Lander GNC equipment summary (right)

Possible improvements to the above equipment summary are:

- RADAR altimeter: Estimated mass (including antennas) ~ 1.7 kg. Power
- Planetary Altimeter breadboard (ABPA) – [TRP Activity] aiming to analyse, design and bread-board two versions (radar and laser) of a compact, low-power, planetary altimeter for usage in ESA exploration (completed in Sep-2018)

34.5.2.4 GNC technology developments

The Planetary Altimeter (continuation of ABPA activity) shall complete its development.

A possible upgrade aiming to reduce the overall mass, is to develop a dual-use LIDAR which could be used both as an altimeter and a hazard-detection-avoidance sensor.

Finally, it should be highlighted that the landing processor unit currently baselined includes a power distribution unit for the camera. It is also estimated that further optimisation of the processor unit could reduce the final weight by more than 50%.

With these modifications, the overall GNC equipment mass (with no redundancy) could be reduced down to 20 kg.

34.5.3 Mechanisms

34.5.3.1 Mechanisms Design Challenges

The main challenge for the Triton Lander Mechanisms subsystem design is the landing gear, in charge of ensuring the following functions:

- Prevent tip over
- Absorb kinetic energy
- Avoid bouncing back
- Ensure ground clearance
- Minimise acceleration (to protect payload).

Other mechanisms would have to be designed (deploying instruments, pointing antenna(s), ...), but this was out of scope of the preliminary assessment of the Lander Concept.

34.5.3.2 Mechanisms Design Drivers

Major design drivers for the Landing gear are:

- Max allowable acceleration (how 'soft' must the landing be)
- Landing speed (this is the energy the landing gear has to absorb)
- Vertical and lateral velocity
- Lander size, CoG
- Clearance needed below lander (payload and terrain would define this)
- Terrain slope.

*Taking conservative assumptions for all the points would lead to oversized solutions or just to infeasibility.

Reasonable assumptions and careful evaluation of reference missions would have to be used in the design.

34.5.3.3 Baseline mechanisms design

The following reference missions have been considered:

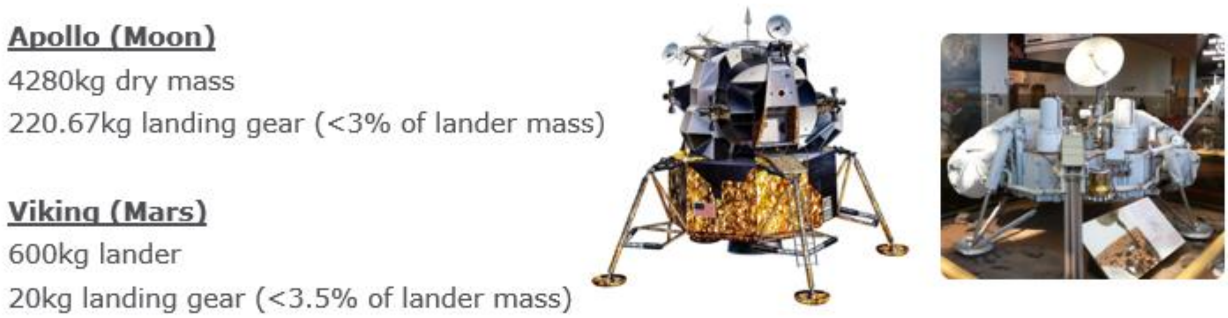


Figure 34-9: Mechanism design reference missions

As a preliminary assumption, landing gears are typically around 3-5% of lander mass. For the Triton case a rough estimate of 20kg (excluding margins) is used as the starting point.

34.5.3.4 Mechanisms technology developments

No peculiar developments in the mechanisms domain are envisaged at this stage.

34.5.4 Power

34.5.4.1 Power trade-offs

A lander power subsystem could be based on:

Battery

- Heritage solution: Lithium primary cell. assuming SAFT LO26SHX $LiSO_2$:
  - 235 Wh/kg BOL nameplate @ cell @ **20°C**.

- x1.3 cells-to-batt mass factor, 13 years x 3% self discharge, 80% DoD, 10% string redundancy.
- 73 Wh/kg EOL total effective usable @ battery level. = **0.33 kg per watt per day**.
- Minimum conceivable power budget: Assuming:
  - 10W PCDU,
  - 20W DHS,
  - 20W AOGNC,
  - 20W COMMS

  TOTAL POWER: 70W
- Flight time: one week (assumption based on considerations over the drift arcs for the orbit before landing. Such drift arcs are driven by several factors:
  - The need for orbit determination in preparation for or following one of the manoeuvres in the sequence (M1 to M4, as explained in the Mission Analysis section)
  - Operational requirements (driven by the orbiter or by orbital dynamics)
  - Surface science in preparation of landing (will observations from the Neptune orbiter gathered during its flybys at Triton be sufficient to characterise the landing site: slope, roughness, other hazards? Does the lander have to do that prior to initiating the final landing sequence?)
  - It can be assumed that as a minimum ~ 1 week is needed and most of these activities require a lander that is very much alive and non-hibernating. At least this is valid for orbit determination and Triton surface observation.
    - Battery mass = 7 days x 70 W x 0.27 kg per watt per day = **162 kg** (excluding SURFACE MISSION, thus is a ''zero science situation")
    - Battery mass (assuming 1 week of Surface mission) = 162 + (=7*(70+50***)*0.27) = 162 + 226.8 kg = **388.8 kg***

    **** 50 W would be the additional required power to transmit data*

- Best case technology solution: Next-generation Li-ion secondary.
  - Assuming:
    - Very low calendar ageing 0.5 %/yr.
    - Top-up charge before release
    - 220 Wh/kg BOL nameplate @ cell @ **20°C**
    - Temperature ensured by RHUs
    - x1.3 cells-to-battery mass factor, 80% DoD, 10% string redundancy
    - 114 Wh/kg EOL total effective usable @ battery level. = **0.21 kg per watt per day**.
  - Minimum conceivable power budget? Assuming:
    - 10W PCDU,
    - 20W DHS,
    - 20W AOGNC,
    - 20W COMMS

TOTAL POWER: 70W

- Flight time: one week
- Battery mass = 7 days x 70 W x 0.27 kg per watt per day = **103 kg** (excluding SURFACE MISSION)
- Battery mass (assuming 1 week of Surface mission) = 103 x 2 = **206 kg**

RTG

- eMMRTG
  - Assuming: 90W EOL.
  - Minimum flight power budget? Assuming:
    - 10W PCDU,
    - 20W DHS,
    - 20W AOGNC,
    - 20W COMMS

    TOTAL POWER: 70W
  - Flight time? As long as needed
  - **45 kg, <u>including</u> surface mission.**

However this solution still needs a small battery in support in case there are any moments exceeding 90W power demand (Propulsion power needs are not considered in the minimum power budget, therefore a battery could be required, unless the timeline is planned in such a way not to exceed the power demand).

PCDU

- Heritage solution: *Huygens* probe
  - 16 kg, plus 5 kg pyro unit = **21 kg**
  - 600 W max power capability
- Best case technology solution: Modern high function-density modular PCDU e.g. Terma *Modular Medium Power Unit*.
  - Approximatively **10 kg** total with all functionalities including pyro actuation

34.5.4.2 Power technology developments

Further mass savings could be expected with a combined avionics/power unit approach, as a "Minavio" concept.

34.5.5 Thermal

34.5.5.1 Thermal design drivers

A Triton Lander will not only require very good radiative decoupling but also an effective conductive decoupling from any contact point on the moon's surface (considering the duration of the operations on ground). Note: Voyager 2 during fly-by found surface temperatures of -235 degC.

34.5.5.2 Thermal baseline design

A preliminary sizing is based on:

- 22-layer MLI using HELPAC spacers from Juice (mass can be estimated based on external surface area)
- Conductive de-coupling somewhat configuration dependent, but likely based on foam or aerogels
- RHU's required already for cruise and coasting (number would be determined by the duration of these phases)
- Radiator windows might be required to reduce sensitivity during coasting
- PCM's might locally be used to protect instruments and units.

34.5.5.3 Thermal technology developments

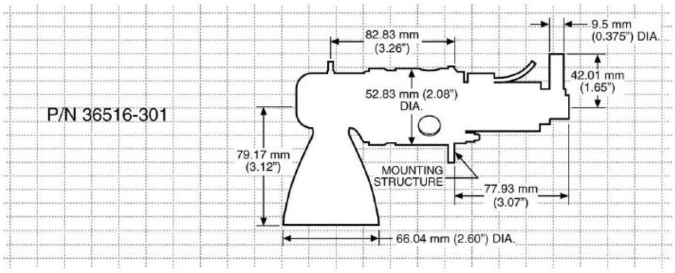
Possibly no technology developments are required, however a well elaborated system testing is required.

34.5.6 Propulsion

The propulsion system is designed to handle the high delta v demands as indicated above as well as the soft landing of the spacecraft. Due to the gravity constant of 0.78m/s² the thrust can be compared to the actual acceleration of the spacecraft on ground. Due to the requirement of the soft landing, a system based on conventional hydrazine thrusters is used. This is due to the fact that those thrusters are due to the blowdown of current systems designed for throttling purposes. The only missing adaptation is a usage of a valve with throttling capabilities. This valve is based on the high pressure proportional valve from Nammo (https://indico.esa.int/event/181/contributions/1375/attachments/1331/1556/Nammo_Electronic_Pressure_Regulator_-_Clean_Space_26-10-17.pdf). This valve could be adapted to be in the direct flow path in hydrazine (technology development needed) and could then throttle the mass flow by having a blowdown system. This enables a regulation of the thrust to compensate the acceleration on the surface.

Due to the large delta v demands, a dual mode system consisting of a main engine and several 220N thrusters using purely hydrazine is baselined. This system consists of one main engine (HIPAT™ DM - http://www.rocket.com/propulsion-systems/bipropellant-rockets) with a nominal thrust of 445N and a specific impulse of 329s for the configuration with an expansion ratio of 375 (http://www.rocket.com/files/aerojet/documents/Capabilities/PDFs/Bipropellant%20Data%20Sheets.pdf). The other thrusters are the MR-107V thrusters from Aerojet as well, characteristics mentioned in the figure below.

MR-107V 220N (49.5-lbf) ROCKET ENGINE ASSEMBLY



Design Characteristics

- Propellant .... Hydrazine
- Catalyst .... S-405 / LCH-202
- Thrust/Steady State .... 220 – 67 N (49.5 – 15 lbf)
- Feed Pressure .... 26 – 5.5 bar (377 – 80 psia)
- Chamber Pressure .... 8.4 – 2.6 bar (122 – 38 psia)
- Expansion Ratio 21.5:1
- Flow Rate .... 98 – 31 g/sec (0.216 – 0.07 lbm/sec)
- Valve .... Single Seat
- Valve Power .... <34.8 Watts @ 28 Vdc & 20°C
- Mass .... 1.01 kg (2.23 lbm)
  - Engine .... 0.67 kg (1.48 lbm)
  - Valve .... 0.34 kg (0.75 lbm)

Performance – *Reference MR-107S/T Testing in 2007*

- Specific Impulse .... 229 – 223 sec (lbf-sec/lbm)
- Total Impulse .... 337,175 N-sec (75,800 lbf-sec)
- Total Pulses 30,275
- Steady State Firing .... 100 sec @ 111 N (25-lbf)

Status

- Qualification Testing in 2007

Rev. Date: 6/19/07
2007-008
11411 139th Place NE • Redmond, WA 98052
(425) 885-5000 FAX (425) 882-5747

AEROJET

Approved for public release and export

Figure 34-10: MR-107V 220N Rocket engine assembly

34.5.6.1 Propulsion system

The propulsion system consists of the engines for landing as well as the main engine for the large delta v manoeuvres.

Description	Type	Amount	Mass per unit	Margin	Mass incl. margin
Pipes	Pipes	1	6	0.2	7.2
AOCS Engines	MR-107V	8	1.01	0.05	8.484
Main Engine	HIPAT-DM	1	5.44	0.05	5.712
Fuel Tank	OST-22X	1	36	0.05	37.8
Oxid Tank	E3000	1	36	0.05	37.8
Fill / Drain Valves		7	0.07	0.05	0.5145
LP Pressure Transducer	SAPT	2	0.216	0.05	0.4536
HP Pressure Transducer	SAPT	1	0.216	0.05	0.2268
Latch Valve		2	0.75	0.05	1.575
Propellant Filter	RA04822A	2	0.077	0.05	0.1617
Check valve	VN005-001	2	0.585	0.05	1.2285
Helium Tank	PVG-120	1	23.5	0.05	24.675
Helium Tank	ATK-80446	0	7.7	0.05	0
Pressure regulator	Vacco	1	1	0.2	1.2

Description	Type	Amount	Mass per unit	Margin	Mass incl. margin
Pyrovalve	Cobham	0	0.315	0.05	0
SMA valve	Arianegroup	2	0.16	0.2	0.384
High pressure latch valve	Vacco V1E10560-01	0	0.8	0.05	0
Proportional valve	Nammo	1	0.5	0.2	0.6
Total					127.0311

Table 34-14: Propulsion system equipment and mass

It was discussed during the session dedicated to the design of the lander whether a possible jettison of a part of the propulsion system could be beneficial. This option was investigated shortly but also has not shown a compliance to the maximum mass requirement.

34.5.6.2 Propulsion manoeuvre baseline

The following scenario for the manoeuvres and the delta v investigations was used:

Manoeuvre	mass begin [kg]	mass end [kg]	velocity increment [m/s]
Manoeuvre 1	2457.6	1261.3	2152.0
Manoeuvre 1 gravity losses	1261.3	1104.0	430.0
AOCS manoeuvre 1	1104.0	1036.3	138.4
Manoeuvre 2	1036.3	949.6	282.0
Manoeuvre 2 gravity losses	949.6	949.6	0.0
AOCS manoeuvre 2	949.6	945.2	10.0
Manoeuvre 3	945.2	943.2	7.0
Manoeuvre 3 gravity losses	943.2	943.2	0.0
AOCS manoeuvre 3	943.2	943.1	0.2
Manoeuvre 4	943.1	532.5	1250
Manoeuvre 4 gravity losses	532.5	532.5	0
AOCS manoeuvre 4	532.5	512.0	85.98146515
Final/Total (Including Residuals)	473.04		4355.6

Table 34-15: Propulsion system manoeuvres

The AOCS manoeuvres are always modelled as a percentage of 5% of the propellant used for the manoeuvre. As an example, the AOCS manoeuvre 1 is 5% of the propellant mass of the manoeuvre 1 and manoeuvre 1 gravity losses. Following table includes the details about the manoeuvres and the propellant mass consumption:

Manoeuvre	Thruster	Mixture ratio	propellant mass [kg]
Manoeuvre 1	HIPAT-DM	1.43	1196.24
Manoeuvre 1 gravity losses	HIPAT-DM	1.43	157.39
AOCS manoeuvre 1	MR-107V	0.00	67.68
Manoeuvre 2	HIPAT-DM	1.43	86.73
Manoeuvre 2 gravity losses	HIPAT-DM	1.43	0.00

Manoeuvre	Thruster	Mixture ratio	propellant mass [kg]
AOCS manoeuvre 2	MR-107V	0.00	4.34
Manoeuvre 3	HIPAT-DM	1.43	2.05
Manoeuvre 3 gravity losses	HIPAT-DM	1.43	0.00
AOCS manoeuvre 3	MR-107V	0.00	0.10
Manoeuvre 4	MR-107V	0.00	410.58
Manoeuvre 4 gravity losses	MR-107V	0.00	0.00
AOCS manoeuvre 4	MR-107V	0.00	20.53
Final/Total (Including Residuals)			1984.55

Table 34-16: Propellant mass consumption by manoeuvre

In total, around 1984.6kg of propellant is needed to land with the MR-107V engine the final mass of 473.04kg on the surface. The relation between thrust and surface acceleration at the end is calculated to be 0.596. Therefore, two engines simultaneous, maybe with an angle in relation to the axis of the spacecraft, are needed to throttle down at the end of the landing. It was as well assumed that a part of the deceleration of the spacecraft can be still done with the HIPAT engine, leading to a maximum thrust of around 800N to land the spacecraft.

34.5.7 Structures & Configuration

34.5.7.1 Structures & configuration design drivers

Major design drivers for the Structures/Configuration design are:

- CoG (in case there is a big battery mass, accommodation shall be carefully addressed)
- Stiffness requirements and loads experienced during landing have to be assessed; however landing on a low-gravity moon makes the structural design less critical. Heritage can be used and Crushable structure should be sufficient, as the lander does not have to take off again after landing
- Need to cant thrusters and reduce plume impingement contamination.

34.5.7.2 Structures baseline design

Structural mass can be assumed to be 20% of the lander dry mass.

The possibility to stop the engine high enough from the surface and then land with a crushable structure (TRL 4 as of today) shall be considered. This could mitigate the contamination effects of the plumes and allow not adopting the sky-crane, as done by NASA (and for which ESA does not have the technology ready).

34.5.7.3 Structures technology developments

No peculiar developments in the Structural domain are envisaged at this stage.

34.5.8 Communication

34.5.8.1 Communication design drivers

Major design driver for the Comms subsystem is the visibility of the orbiter in order to be able to relay scientific data.

34.5.8.2 Communication baseline design

Due to the maturity level of the payload timeline definition and the required data volumes, it is assumed to adopt the Neptune Probe comms baseline design, with an RF power of 20W, derived from a 70 peak power consumption for comms.

Under the assumption that visibility of the orbiter is ensured, then the bitrate shall be selected as a function of the maximum distance from the probe as follows:

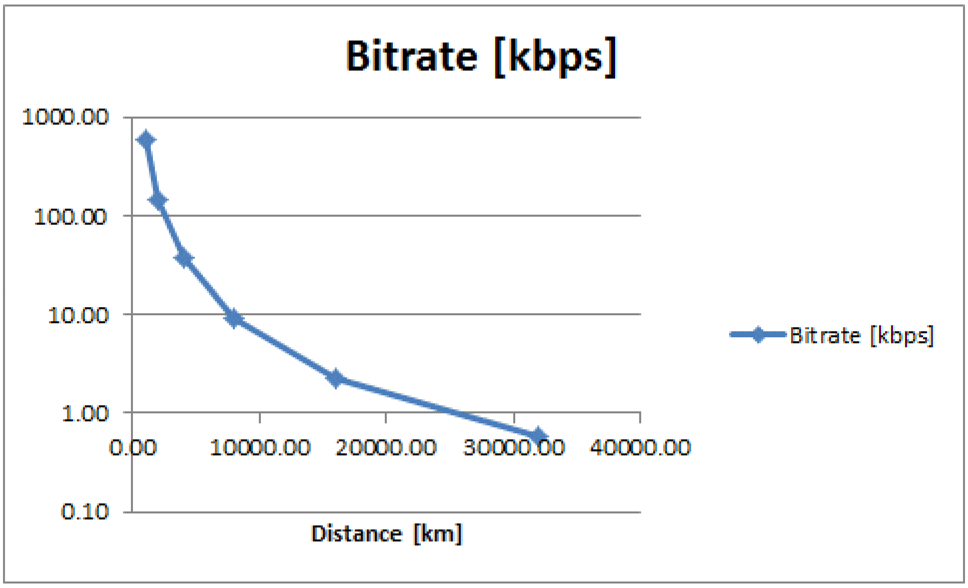


Figure 34-11: Bitrate versus distance to orbiter

Source Parameters of the link budget computation are as follows:

PARAMETER	Value	Notes
RANGE [km]	40000.0	
FREQUENCY [MHz]	450	
TX POWER [W]	19.5	**Power assumption**
TX ANTENNA GAIN [dB]	-0.63	LGA
TX LOSSES [dB]	1	
TX EIRP [dBW]	**11.27**	**Calculated**
PATH LOSSES [dB]	**177.55**	**Calculated**
ATMOSPHERE LOSS [dB]	10.65	Assumption for Triton
RX G/T [dBK]	-19.00	Assuming 14.5 dB gain
DEMOD. LOSS [dB]	1.00	Estimation
MOD. LOSS [dB]	1.24	Suppressed carrier modulation
REQUIRED Eb/No [dB]	1.80	LDPC, Proximity-1
MINIMUM MARGIN [dB]	3.00	Standard ESA
MAX BIT RATE [dBHz]	25.64	
MAX BIT RATE [kbps]	**0.37**	

Table 34-17: Link budget

34.5.8.3 Communications technology developments

No peculiar developments in the Communication domain are envisaged at this stage.

34.5.9 Operations

34.5.9.1 Operations design drivers

- Major design drivers for the Operations are:
- AOCS complexity (level of autonomy)
- Availability of the communication link to ground or to orbiter
- Visibility from Earth.

34.5.9.2 Operations Baseline Design

An array of two 35 meters antennas is the baseline, as for the Neptune probe and orbiter.

34.5.9.3 Operations technology developments

There is the idea to develop an array of up to 4 ground stations, with 2 receiver-only antennas, and 2 Transmitting + Receiving (less costly)

34.5.10 Risks

The major criticalities identified at this stage, and to be properly addressed, mitigated and controlled, are the landing hazards:

- Stability
- Landing in “fluffy” ice with consequent loss of communication
- Plume contamination

34.6 Technology Needs

Included in this table are:

- Technologies to be (further) developed
- Technologies identified as coming from outside ESA member states
- Technologies available within European non-space sector(s)

	Technology Needs					
*	Equipment Name & Text Reference	Technology	Supplier (Country)	TRL	Funded by	Additional Information
	PROPULSION	Throttled / pulsed propulsion capabilities in a closed-loop GNC system for the final descent manoeuvre				

	GNC	Dual-use LIDAR, hazard-detection-avoidance and altimeter.				
	POWER	Combined avionics/power unit approach, as a "Minavio" concept.				
	OPS	Array of up to 4 ground stations, with 2 receiver-only antennas, and 2 Transmitting + receiving				

34.7 Europa Lander Mission

This section provides a summary of the current status of the NASA mission Europa Lander, as understood from several sources available on the internet. It must be stressed that the mission is still going through multiple iterations and it is not easy to ensure that all the inputs found in the mentioned references are fully coherent among themselves. Nevertheless the following summary is believed to be sufficiently representative of the mission design and provides a good sample case to compare to the Triton Lander scenario.

The Europa Lander mission will use Space Launch System (SLS) as launch vehicle in Dec-2025 and arrive to Jupiter in Jul-2030 by means of a propulsive Earth gravity assist. Another gravity assist manoeuvre, this time at Ganymede, is planned prior to the Jupiter Orbit Insertion (JOI) and a series of gravity assists manoeuvres at Ganymede and Calisto will result in a Jupiter tour of 18 months from JOI to Europa encounter. After the lander release, the carrier will execute several transfer orbits to reach a final relay orbit, allowing more than 10 hours communication coverage with the lander (RD[55]).

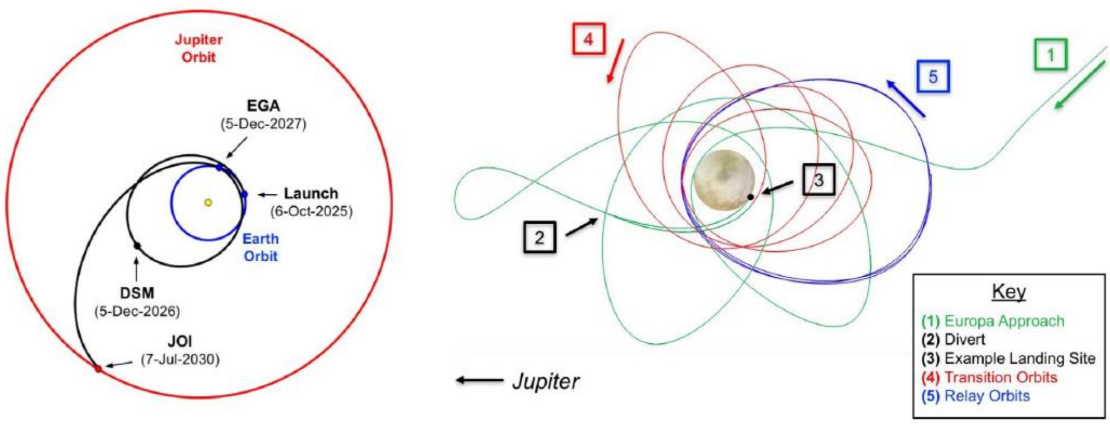


Figure 34-12: 2025 launch with a 4.8 year transfer (left) and Carrier/Lander trajectories around Europa (right)

This trajectory is designed to reduce the spacecraft velocity relative to Europa. After the Mission Concept Review in June 2017, NASA explored several concepts to reduce cost (RD[56], RD[59]): abandon the orbiter in a stable orbit after separation and ensuring direct Earth communication of the lander or extending Europa Clipper mission by 3 years (to be launched in 2022) and use it as satellite relay.

All these options impact on the lander velocity relative to Europa, which is the main driver for the lander design. For the option using the carrier as orbiter relay, the lander velocity relative to Europa at 6 km periapsis altitude is equal to 1850 m/s (RD[57]), while for the case of Europa Clipper as satellite relay the entry velocity can increase up to 2300 – 2900 m/s (see page 136 of RD[56]).

The resulting spacecraft design is shown in Figure 34-13, taken from RD[56], page 129.

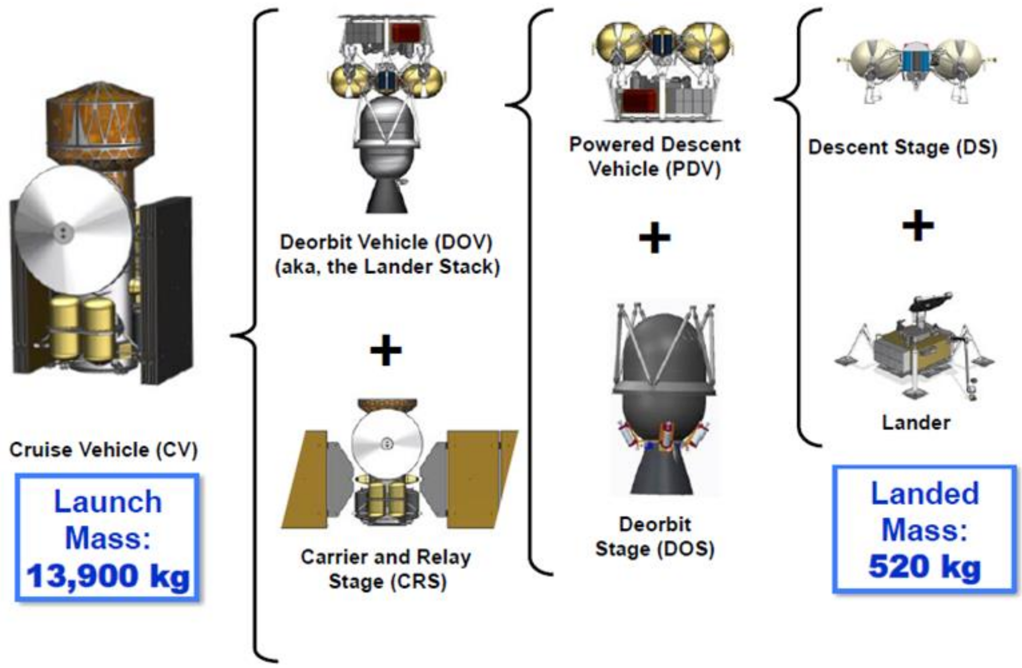


Figure 34-13: Europa Lander Flight System (2/3 of the total mass is devoted to propulsion needs). Note how the lander stack is covered by a bio-shield at lift-off

The solution proposed by NASA uses a two-stage propulsion system followed by a sky-crane landing. The main delta-v manoeuvre is performed by a fixed-nozzle solid rocket motor (SRM) Star 48 class. The attitude control during the thrust of the SRM is done by 4 hydrazine engines MR-104 G (800 N each). The burn lasts approximately 60 seconds, and afterward the SRM is jettisoned. The descent stage is now on charge of reducing the remaining 100 m/s velocity down to 0 at touch-down.

Based on all the available information, Table 34-18 shows an estimation of the overall mass budget of the Europa Lander Flight System, together with the thrust level and delta-v of each propulsion module assuming a Europa arrival velocity of 1750 m/s.

	mass [kg]	dv [m/s]	Isp [s]	Thrust [KN]
Carrier and Relay Stage (CRS)	**10717**			
CRS dry mass	2559			
CRS propellant	8158	2602	300	?
Deorbit Stage (DOS)	**2091**			
DOS dry mass	591			
DOS propellant	1500	1750	280	70
Descent Stage (DS)	**572**			
DS dry mass	500			
DS propellant	72	200	300	3.2
Europa Lander	**520**			
Cruise Vehicle (CV)	**13900**			
CV Total dry mass	4170			
CV Total propellant	9730			

Table 34-18: Europa Lander Flight System estimated mass budget

No information was found about the Descent Stage dry mass, so it has been assumed to be 500 kg. RD[57] states that the DS is responsible for braking the last 100 m/s of velocity. A 100% margin has been added to take into account hovering during sky-crane operations and re-targeting during the hazard avoidance manoeuvre. Those 200 m/s delta-v result in 72 kg of DS propellant. RD[57] also states that the deorbit stage (DOS) delivers a delta-v of 1750 m/s burning 1500 kg in approximately 60 seconds. With that information, the dry mass of the DOS has been computed as 591. The propellant mass of the Carrier and Relay stage has been derived imposing that the total propellant mass of the mission is 2/3 of the total mass (as stated in RD[56])

Due to the uncertainties of the SRM burn duration, the position error is approximately +/- 4 km. To reduce this error, the Europa Lander mission will perform absolute visual navigation, comparing the images taking by an on-board camera with the reconnaissance maps provided by Clipper. The lander will then select a safe landing area using hazard detection & avoidance algorithms based on LIDAR and camera images. The guidance manoeuvre towards the safe landing site is performed with another 4 mono propellant MR-104 G engines, canted 5 deg. At an altitude of approximately 24 m altitude, the lander and the Descent Stage will separate and the sky-crane will deploy a 13 m cable, enabling a safe landing. During the sky-crane operation, a different set of 4 MR-104 G engines is used. These are canted 30 deg (to prevent landing site contamination) and are throttled down to 30% to allow a thrust to weight ratio close to 1.

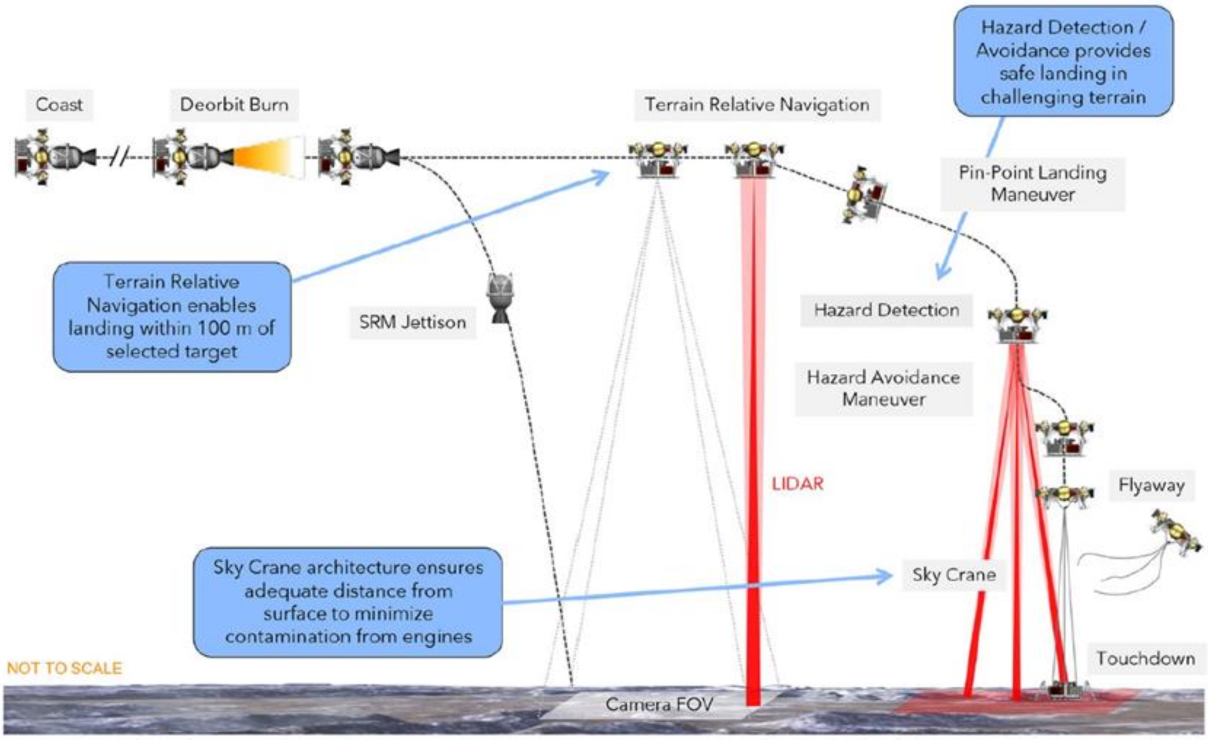


Figure 34-14: De-orbit, Descent and Landing sequence of Europa Lander RD[55]

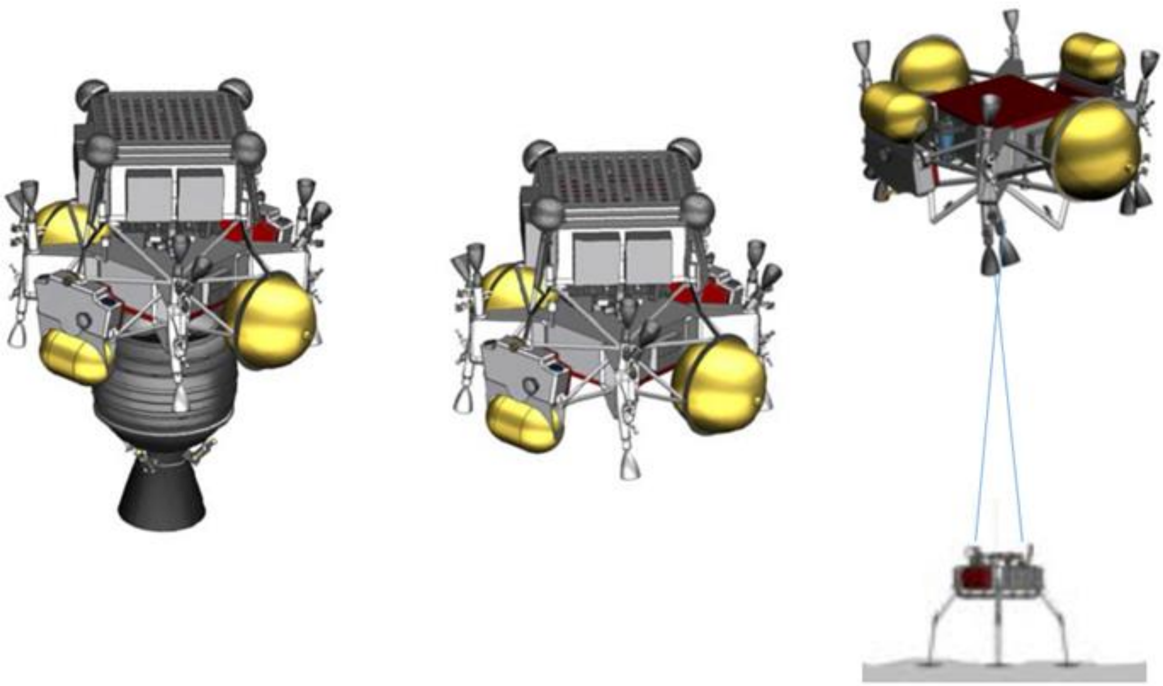


Figure 34-15: De-orbit (left), Power Descent (center) and Sky-crane configurations for the Europa Lander mission RD[55]RD[61]

Based on the above literature survey, there are several key aspects from the Europa Lander mission which can be relevant for a Triton lander mission:

- Two thirds of the total mass of the system is devoted to propulsion needs
- Bringing a lander in the vicinity of Europa at a reasonable small velocity (1750 - 2900 m/s) requires a dedicated mission of 14 metric tons for a lander mass of roughly 500 kg
- A dedicated carrier delivers the lander stack near Europa such that the lander stack performs only one major delta-v manoeuvre prior to land. The carrier is either discarded after that or used shortly as relay satellite for the lander, but it is not expected to do additional science
- The GNC assumes reconnaissance imaginary created by a previous mission (Clipper) to help of the selection of a safe landing area. The final hazard detection and avoidance algorithms do not need prior knowledge of the terrain and are based on LIDAR and camera images
- One of the key challenges is the compliance to the requirements of a planetary mission classification Category IV, which implies that the whole lander stack shall be covered with a bio shield (less than 1×10^{-4} probability of contaminating the European ocean by a viable Earth micro-organism). (RD[55])
- In order to avoid landing site contamination during landing, the lander uses a large sky-crane (> 13 m cable) with highly canted braking engines (30 deg)
- The landing system uses long legs at touch down (> 1.5 m length), most likely to be able to cope with very irregular terrain. Once the vehicle is stabilised, the legs can be folded during surface operations.

35 RADIATION

35.1 Requirements and Design Drivers

The requirements that drive the radiation design are derived from the expected effects potentially experienced by the mission from the radiation environment sources.

In the ESA PEP CDF study (RD[7]), requirements from potential radiation effects were not considered. The NASA Ice Giants Report (RD[64]) includes a generic reference to a radiation total dose level of 30 krad (behind a 100 mil thick aluminium shielding (~2.5 mm), with an RDM of 2 added), probably only including the effect of Solar energetic particles (SEPs) in Appendix B (A-Team Study Report from the Ice Giants Workshop).

Use of RTGs, Jupiter flybys and extended duration of the orbiter mission segments at the destination planets require increased attention to the constraints imposed by radiation.

35.1.1 Design Drivers: Radiation Effects and Main Sources of Radiation Environment

Radiation effects can in general be categorised as:

- Cumulative degradation effects from total ionising dose (TID) or total non-ionising dose (TNID)
- Single event effects (SEEs) from single particles.

The main radiation sources to be included in a complete radiation analysis for the mission, along the mission timeline, are summarised in Table 35-1. In this study, SEEs and TNID are not considered, and TID is included as main design driver.

Radiation sources			
Source	**Particles**	**Time/Location**	**Impact**
Solar Particle Events (SPEs)	Solar energetic particles (SEPs)	During the entire mission (13 year transfer and 2 years at the destination planet)	Short term high fluxes, contribution to SEEs and cumulative doses
Galactic Cosmic Rays (GCR)	Protons and heavier ions	During the entire mission (13 year transfer and 2 years at the destination planet)	Continuous low intensity flux, contribution to SEEs, only limited cumulative dose
Radioisotope thermoelectric generators (RTGs)	Neutrons and gammas	Local artificial environment, during the entire mission (13 year transfer and 2 years at the destination planet)	Continuous low intensity flux, contribution to TNID/TID, potentially SEEs
Jovian trapped energetic particle environment	Energetic electrons and protons	In the vicinity of Jupiter	Short term high intensity fluxes, contribution to total mission cumulative doses and SEEs
Local planetary trapped radiation	Electrons and protons	2 years at the destination planet	Potential short term fluxes. contribution SEEs and total cumulative doses

Table 35-1: Main sources of radiation considered in the study

35.2 Assumptions and Trade-Offs

This Section includes the assumptions made for the sources of radiation included in the analysis for the CDF study. For each source, the method used to compute the source intensity and the related TID level is outlined, together with potential uncertainties and related proposed margins.

35.2.1 Solar Particle Events (SPEs)

Solar energetic particles (SEPs) are emitted from the Sun during Solar Particle Events (SPEs), often categorised into Solar flares or Coronal Mass Ejections (CMEs), both linked to reconfiguration of solar magnetic field. The CMEs, responsible for the greater disturbances, are event driven phenomena, with occasional high fluxes over short periods. Solar particle species include protons, some ions, electrons, neutrons, gamma rays, X-rays, but the dominant contribution to TID and TNID comes from the protons.

The total mission fluences computed for the mission (worst case transfer to Neptune: 13 year transfer and 2 years at the destination planet) are based on the probabilistic ESP model (RD[7]),prescribed by ECSS standards (RD[67]), with 11 years of Solar maximum and 90% Confidence Level (CL).

Flux scaling could be applied in principle to the SPE fluences to take account of the varying Helio-radial distance along the mission trajectory. However, given the statistical nature of the SPE occurrence and the significant uncertainties in the flux variation, the conservative approach is adopted of computing the total particle fluence at 1 AU.

The TID calculations are based on SHIELDOSE-2 (RD[76]) (solid-sphere geometry case).

Margin: The fluence estimates from ESP are based on a Confidence Level-based risk approach; therefore, no extra margin is required to cover the stochastic nature of the Solar event phenomena. Potential systematic uncertainties in the ESP model are being analysed in ongoing ESA-led projects and are addressed in new related model developments (e.g. SAPPHIRE, RD[66]) but are not taken into account in this study.

35.2.2 Radioisotope Thermoelectric Generators (eMMRTGs)

The eMMRTGs create an additional local artificial radiation environment, during the entire mission (13 year transfer and 2 years at the destination planet). The radioactive decay chain in the RTGs pellets induces emission of neutrons and high energy gamma rays, which can easily traverse the structures of the RTG and of the rest of the spacecraft, and reach sensitive devices of platform and payload.

The resulting continuous, low intensity flux of neutrons and gammas contribute respectively to TNID and TID, and potentially SEEs. Estimates in this study for TID and TNID (expressed in equivalent 1 MeV neutron fluence) are based on dose maps from NASA Ice Giants report RD[64], reported in Figure 35-1.

Based on example spacecraft configurations in the NASA report, such as the one reported in Figure 35-2, it is here conservatively assumed that sensitive electronic devices are at 1m distance from the 1st RTG, at 1.5m distance from the 2nd RTG, and that other RTGs are more distant and can therefore be neglected. RTG particle emission decrease during the mission duration is neglected, and no reduction in TID or TNID levels is assumed from the spacecraft structures due to shielding. The resulting dose levels for the total 15 year duration of the mission are reported in Table 35-2. It is assumed that the eMMRTGs are well characterised and no additional margin factor is included in this study for the dose levels.

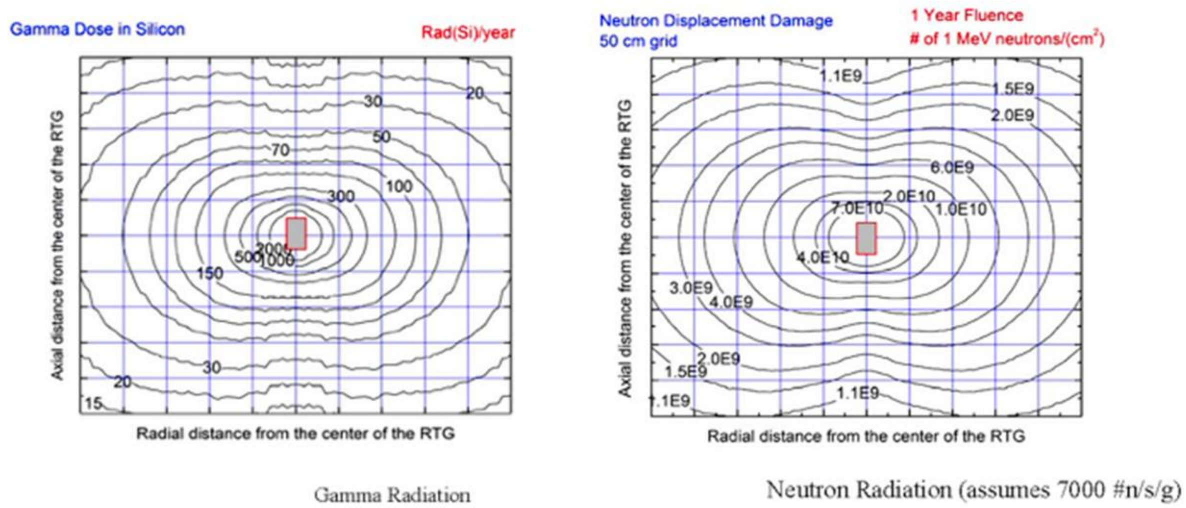


Figure 35-1: TID and TNID (expressed in equivalent 1 MeV neutron fluence) dose maps from NASA Ice Giants report ([])

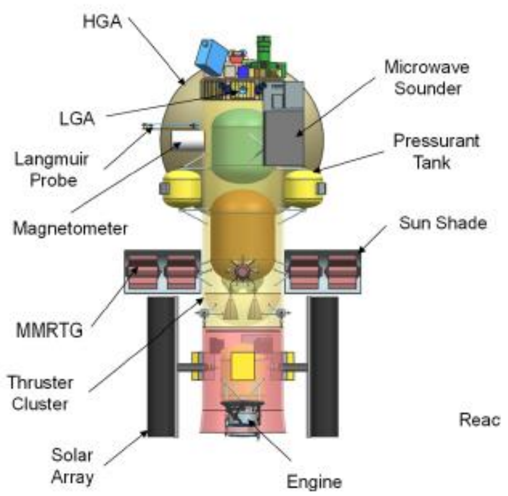


Figure 35-2: Example configuration from the NASA Ice Giants report (RD[64])

Distance [m]	gamma TID [krad]	neutron (TNID) eq fluence [1MeVn/cm^2]
0.5	12.0	9E+11
1.0	3.8	2E+11
1.5	1.4	1E+11
2.0	0.8	5E+10

Table 35-2: TID and TNID dose levels induced by the RTGs for the entire mission duration

35.2.3 Jovian Trapped Energetic Particle Environment

The Jupiter magnetic field traps electrons, protons, and heavier ions in Van Allen belts around the planet. The intensity of the field and size of the magnetosphere allow for trapped electrons with energies probably in excess of 100 MeV, significantly higher than those observed in the Earth trapped environment, where typical electron energies are lower than 10 MeV. Jupiter rotational period of 9 h 56 min makes plasma torus and radiation belts wobble due to a 7° tilt between Jupiter rotational and magnetic axes.

Several models have been developed at JPL in the past decades to describe the Jovian trapped particle environment, starting with the Divine&Garrett model (1983, RD[68]), and continuing with its updates: GIRE (2003, RD[69] RD[70]), and GIRE2 (2012, RD[71]).

The predictions in this study are based on the JOSE model RD[72], developed in the context of the ESA-led JORE^{2}M^2 project, in preparation for the ESA JUICE mission, and used for the mission design. Close to Jupiter, for a magnetic shell parameter L<9.5, JOSE internally makes use of the ONERA Salammbo model (RD[73]). For distances from Jupiter beyond the extension of JOSE, the Interplanetary Electron Model (IEM, RD[74]) is used. All calculations are performed with the models as available in the SPENVIS system.

The calculations for both the Neptune and the Uranus case are based on the detailed trajectories produced by the ESA/ESOC Mission Analysis team.

- The Neptune Jupiter segment follows a hyperbolic trajectory with 875,000 km minimum altitude. The trajectory is near-equatorial (see Figure 35-3 left), and therefore encounters the more intense regions of the radiation environment, although at relatively big distance from the planet. Although the full trajectory segment lasts 120 days, the spacecraft is expected to traverse the most intense radiation regions in just a few days.
- The Uranus case is rather different from the Neptune one: the flyby has a closest approach altitude of just 10,000 km. In a near-equatorial case, this would bring the spacecraft through the extreme intensities of the core of the Jupiter trapped electron belt. Instead, the trajectory has a very high inclination (see Figure 35-3 right), allowing for a significant reduction in dose levels.

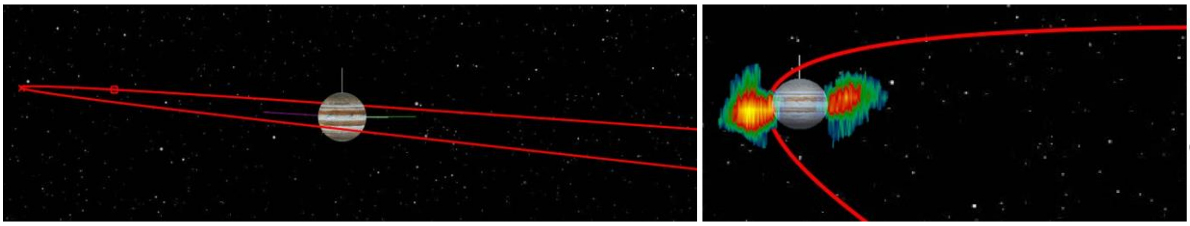


Figure 35-3: Graphical representation of the Jupiter flyby mission segment for the Neptune case (left) and the Uranus case (right)

The TID calculations are based on the SHIELDOSE-2Q RD[75] code (an extension of SHIELDOSE-2 RD[76]) for the solid-sphere geometry case.

There are significant uncertainties in the estimates of Jupiter trapped energetic electrons flux levels. These uncertainties have two aspects:

- Systematics may affect the models due to both the limited dataset upon which the prediction models are based, and to errors in the measurements themselves (for example signal saturation, as visible in the EPD data in the left plot of Figure 35-4, as analysed by Evans in RD[77]).

- Significant variations in the particle flux time series have been observed in the available dataset for the same region in the Jovian system. See for example the wide variations in the scattered data in the EPD data plots in Figure 35-4 for both the lower energies (left) and the higher ones (right). Predictions of averages over long periods are less affected by the short-term variations, than those over short duration trajectories such as in the case of a single flyby.

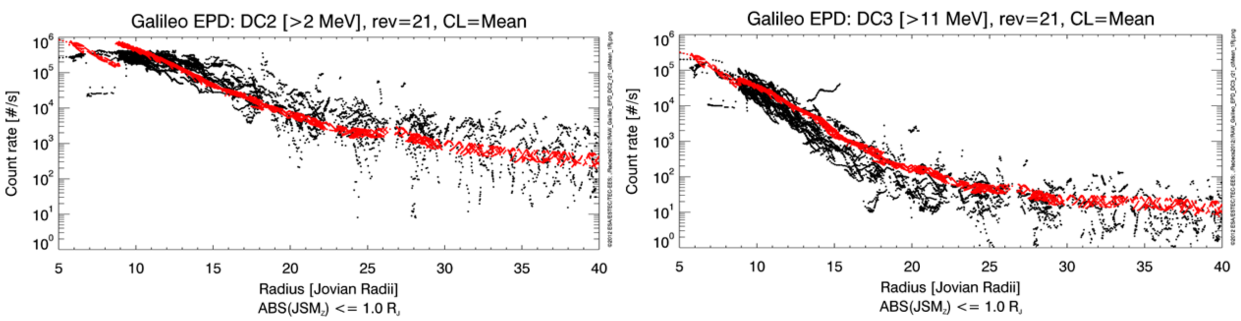


Figure 35-4: Galileo EPD DC2(>2MeV) and DC3 (>11 MeV) data as analysed by Evans (RD[77])

In order to take account of these aspects, the environment margin strategy adopted for the ESA JUICE mission includes:

- A factor x2 to be applied to long term average electron fluences (for TID, TNID)
- A factor x4 to be applied to the worst case fluxes (e.g. for instrument background or short term SEE analyses).

The approach for the Ice Giants study flybys needs reconsideration of the margin policy to take account of important differences with respect to the JUICE mission.

For the Neptune case:

- The flybys have a much shorter time-scale than the total JUICE trajectory segment in the Jovian system. Therefore, the full range of flux oscillation needs to be considered
- Available Galileo data suggest bigger electron flux variations in time at distance from Jupiter larger than Europa or Ganymede orbits.

As a result, the margin that has been applied to the TID predictions for the Jupiter flyby for the Neptune case is a factor 5x.

For the Uranus case:

- Available data seem to indicate that flux variations closer to Jupiter (e.g. at Europa distance, or at 10k km altitude) are probably smaller than at the Jovian distances typical of the JUICE mission (e.g. Ganymede ~15 RJ). The trapped population appears more stable at low altitudes.

- At lower Jupiter altitudes, the Galileo EPD dataset, which is the main source of data for all models, show a suspected data saturation (see e.g. the DC2 channel for E>2MeV)
- For the trajectory segments inside the Europa orbit, modelling heavily depends on the original Salammbo Jupiter physical model, whose validation is very challenging, giving the limited dataset, implying potential significant model systematics. The 2018 update of this model (RD[77]) could not be used for the study.

As a result, the margin that has been applied to the TID predictions for the Jupiter flyby for the Uranus case is a factor 3x.

35.2.4 Local Planetary Trapped Radiation

Both Neptune and Uranus have a global magnetic field that keep particles trapped in Van Allen belts. There is no standard model for trapped radiation at the two planets, and the development of European models would enable raising the confidence in the dose and flux level predictions. Extremely limited measurement datasets for model construction and validation should also be augmented with new in-situ data by flying radiation monitors on board orbiters at both planets.

35.2.4.1 Neptune

Mechanisms for trapping, acceleration, losses for electrons and protons in the Neptune belt are not very well known. The highly tilted magnetic field with respect to the planet rotation axis is believed to induce a complex interaction with heliosphere.

Significant uncertainties are associated to Neptune trapped energetic electrons and proton flux levels due to the lack of a comprehensive model for trapped radiation. A recent effort by JPL led to the creation of the Neptune Radiation Model (NMOD, RD[79]), covering electrons and protons between 0.025 MeV and 5 MeV based on the California Institute of Technology's Cosmic Ray Subsystem and the Applied Physics Laboratory's Low Energy Charged Particle Detector from the Voyager 2 Neptune flyby of 1986.

The JPL NMOD model is not available for use at ESA, although there has been some exchange of information with JPL and the model development report is available. Earlier publications and the NMOD report allow us to predict in general lower energies, lower fluxes for Neptune orbiter than in typical Earth orbits.

The estimates of radiation exposure at the planet are based on the conservative assumptions of:

- a 2y long Neptune phase consisting of 10d duration orbits, with 1 day of exposure per orbit, leading to a 0.2 year exposure
- a fixed dose rate of 2E-04 rad/s behind 100 mil (~2.5mm) of aluminium shielding (maximum from Section 10 of the NMOD report).

The proposed uncertainty margin to be applied to the TID levels is a factor x5, justified by the limited validation and the limited energy coverage of the model, only including the lower energy range.

Related proposals from this study, aimed at overcoming these limitations, include:

- Initiation of R&D for development of European models for radiation/ plasma/ dust at Neptune
- Embarkation on the Neptune orbiter of a low mass, low power radiation monitor extending to species and energies relevant for dose and SEE predictions.

35.2.4.2 Uranus

As for the Neptune case, also the Uranus trapped environment is heavily affected by its interaction with the heliosphere, due to the large tilt (of about 58 degrees) between its magnetic and rotation axes.

Significant uncertainties affect our knowledge of the trapped electrons and proton flux levels. The Voyager 2 flyby data is currently the only in-situ measurement dataset for model development and validation. A first radiation model had been developed for electrons in the range 0.7 – 2.5 MeV by Selesnick and Stone (1991, RD[80]) based on Voyager 2 data from the Cosmic Ray Subsystem TET electron telescope. The JPL Uranian Radiation Model (UMOD, RD[81]) extends the electron range down to 0.022 MeV, and additionally includes protons in the range 0.028 - 3.5 MeV, by using data taken during the Voyager 2 flyby by the APL Low Energy Charged Particle detector (LECP). These models are not available at ESA for calculations for this study, and no ESA model has been developed yet in preparation for missions to Uranus.

In terms of spectra the Uranus trapped environment is assumed to be qualitatively similar to the Earth one, although maybe lower in intensity. The particle higher energy ranges, of relevance for radiation analyses, i.e. allowing for sufficient penetration through the spacecraft structures, are not included in the aforementioned JPL model, in particular for protons. However, there is indication that TID levels for orbits in the Uranus Van Allen belts cannot be neglected. An analysis by Mauk et al. (2010, RD[82]) comparing the trapped planetary environments, suggests that electrons at Uranus may have an energy spectrum similar to the Earth trapped one. A plot from RD[82] is reported in Figure 35-5, including high intensity spectra for Jupiter, Neptune, Uranus and Earth.

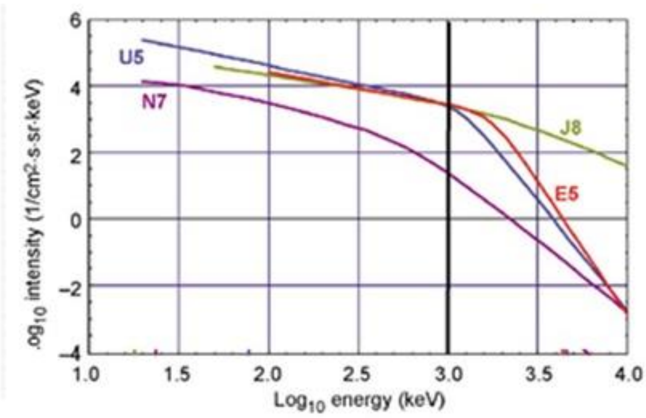


Figure 35-5: Planetary trapped radiation levels at most intense location, from Mauk's 2010 paper (RD[82])

The estimates of radiation exposure at the planet are based on the assumptions of:

- A 2y long Uranus phase consisting of 10 day duration orbits, with 1 day of exposure per orbit, leading to a 0.2 year exposure
- A fixed dose rate, computed with SHIELDOSE-2 (solid sphere) based on the planet most intense predicted spectrum in Mauk's plot (RD[82]), by using its spectral shape (formula 7 in the paper) with the parameters related to Uranus (Table 1 in the paper).

The exposure duration is considered conservative, and dose rate from Mauk's paper represent the predicted trapped radiation levels at most intense location. As a consequence, it is not deemed necessary to apply further margin to the TID calculation. A comprehensive radiation model extending to higher energies for both electrons and protons would allow to follow a less conservative, probably more realistic calculation approach, potentially reducing the TID estimates.

Related proposals from this study, aimed at overcoming the limitations, include:

- Initiation of R&D for development of European models for radiation/ plasma/ dust at Uranus
- Embarkation on the Uranus orbiter of a low mass, low power radiation monitor extending to species and energies relevant for dose and SEE predictions.

35.3 Baseline Design

This Section gives a summary of the radiation doses for the Neptune and for the Uranus cases. Table 35-3 and Table 35-4 include the combined End of Mission TID estimation from all the sources outlined above, respectively for the Neptune and Uranus case.

Each individual source is including its own specific margin, as explained and justified in the previous sections, therefore no additional environment related margin is required. However, an additional margin factor of x1.2 (20%) is added to the total TID values, accounting for the remaining Radiation Hardness Assurance (RHA) uncertainties.

Neptune							
	TIDL (no factors) [krad]			Uncertainty factor	**TIDL [krad]**		
Shielding (mm Al)	2.5 mm (~100 mil)	4 mm	10 mm		**2.5 mm (~100 mil)**	**4 mm**	**10 mm**
SEPs p (interplanetary 15y)	31	18	6	1	31	18	6
RTGs (gamma, @1m+1.5m, 15y)	5	5	5	1	5	5	5
Jupiter e- (flyby 875k km)	14	6	1	5	69	32	7
Planet trapped (JPL NMOD 0.2y @max)	1.3			5	6		
Total					**111**	**55**	**18**
Total including 1.2 RHA margin					**133**	**66**	**21**

Table 35-3: Combined End of Mission TID estimation from all sources for the Neptune case

Uranus							
	TIDL (no factors) [krad]			Uncertainty factor	TIDL [krad]		
Shielding (mm Al)	2.5 mm (~100 mil)	4 mm	10 mm		**2.5 mm (~100 mil)**	**4 mm**	**10 mm**
Interplanetary transit	31	18	6	1	31	18	6
RTGs (gamma, @1m+2m, 15y)	5	5	5	1	5	5	5
Jupiter flyby detailed trajectory	17	12	7	3	51	36	21
Planet trapped (Mauk WC spectra 0.2y)	855	70	2	1	855	70	2
Total					942	129	33
Total including 1.2 RHA margin					**1130**	**155**	**40**

Table 35-4: Combined End of Mission TID estimation from all sources for the Uranus case

A 4 mm Al shielding thickness is a reasonable, easily achievable level of protection (including S/C, Unit, board elements) without excessive additional shielding structures. As a consequence, the reference TID levels resulting from these analyses are 66 krad for the Neptune case, and 155 krad for the Uranus case (including margins).

As a side note, had Jupiter flyby at 10k km minimum distance been near-equatorial (as for the Neptune case), the TID predicted by the JOSE model for the Jupiter flyby alone would be 366 krad(Si) behind 4 mm Al (incl. x3 margin).

At the beginning of the CDF Study the radiation analysis run on preliminary trajectories (referred to as "Original Analysis" in the Uranus System Chapter) produced worse results. A sensitivity analysis highlighting the shielding mass penalty, and delta qualification required by sensitive electronic equipment, implied by the most severe radiation environment is reported in the System chapter.

Uranus							
	TIDL (no factors)			Uncertainty factor	**TIDL**		
	[krad]				**[krad]**		
Shielding (Al)	2.5 mm (~100mil)	4 mm	10 mm		**2.5 mm (~100mil)**	**4 mm**	**10mm**
Interplanetary transit	31	18	6	1	31	18	6
RTGs (gamma, @1m+2m, 15y)	5	5	5	1	5	5	5
Jupiter flyby 10k km	221	122	42	3	662	366	125
Planet trapped (JPL UMOD)							
Planet trapped (Mauk WC spectra 0.2y/20)*	42.8	3.5	0.1	10	428	35	1
Planet trapped (Earth 2y/5)	7	2	0	10	71	19	1

Uranus							
	TIDL (no factors)			Uncertainty factor	**TIDL**		
	[krad]				**[krad]**		
Shielding (Al)	2.5 mm (~100mil)	4 mm	10 mm		**2.5 mm (~100mil)**	**4 mm**	**10mm**
Total					**1125**	**424**	**137**
Total including 1.2 RHA margin					**1350**	**509**	**164**
*For Earth, TID from Mauk's spectra is ~20x MEO							

Table 35-5: Combined End of Mission TID estimation from all sources for the Uranus case

35.4 List of Equipment

The Radiation sub-system includes as equipment the radiation monitors necessary for the in-situ measurement of radiation environment.

Neptune case: Radiation Monitor: currently represented in the mission study configuration by the ESA Next Generation Radiation Monitor (NGRM).

Uranus case: Radiation Monitor:(Radiation Hard) – Currently represented in the mission study configuration by the ESA RADEM monitor being developed for the ESA JUICE mission.

35.5 Technology Needs

Technology Needs						
*****	**Equipment Name & Text Reference**	**Technology**	**Supplier (Country)**	**TRL**	**Funded by**	**Additional Information**
*	Radiation Monitor	Low mass, Radiation Hard				Can be based on the RADEM radiation monitor being developed for the ESA JUICE mission
*	Radiation Monitor	Low mass				Can be based on the ESA Next Generation Radiation Monitor (NGRM)
	Neptune and Uranus models for trapped radiation, plasma, micrometeoroids	Prediction models for generic mission trajectories				Based on (limited) measurement datasets of radiation and other knowledge (magnetic field, etc)

* Tick if technology is baselined

36 GROUND SEGMENT AND OPERATIONS

36.1 Requirements and Design Drivers

The following lessons learned from the Rosetta mission are recommended to be included as mission requirements for the Ice Giants mission:

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
GS-010	Margins for propellant budget allocated for stochastic navigation shall be retained for in flight use.	RD[91] ID-01
GS-020	Landers shall not be treated as payloads/scientific instruments. ESA to be involved in the design and in the operations of the lander type missions.	RD[91] ID-02
GS-030	The dependency on scientific instruments and products for operational purposes shall be minimised.	RD[91] ID-03
GS-040	The spacecraft design (spacecraft and operations) shall be such that it is possible to recover all TM stored during cruise phases. There shall be the capability to recover additional TM ad hoc, e.g. when an anomaly occurs.	RD[91] ID-06
GS-050	For missions having to use optical navigation and dynamic trajectory planning, the processes of navigation, trajectory and attitude planning shall be done by the same entity.	RD[91] ID-07
GS-060	OBCPs shall be available to the mission. ESA shall be involved in early phases of the OBCP specification, development and testing.	RD[91] ID-10

The main design drivers for ground segment and operations identified during the CDF study are:

- NASA <– >ESA Project. NASA leading with ESA partnering. Mission is developed by NASA with ESA proposing and implementing their contribution upon agreements. Possible contributions are either a Neptune Orbiter or Uranus Orbiter launched on a dual launch together with a NASA orbiter. Or a Probe carried by a NASA orbiter to either Neptune or Uranus.
- Orbiter and Probe designs depend on use of RHUs and RTGs
- Orbiter concepts address multiple options driving definition of different transfer and target arrival operational baselines.
- The probe lifetime is for a direct entry descent in Earth visibility of 90 minutes, with HK and science TM data transferred to the NASA carrier which serves as the relay to Earth.

36.2 Assumptions and Trade-Offs

36.2.1 Assumptions

Assumptions	
1	The Ice Giants part of the mission is conducted by ESA, reliant on ESA ESOC/ESAC infrastructure.
2	The ground segment and operations infrastructure for the Mission Operations Centre (MOC), of the Ice Giants mission (Orbiter or Probe) will be set up by ESA ESOC, and will be based on an extension of the ESOC ground segment infrastructure customised to meet the mission specific requirements.
3	The Ice Giants mission (Orbiter or Probe) will be operated from the MOC at ESA ESOC, by the Flight Control Team that forms part of the Solar System & Exploration Missions Division under OPS-OP.
4	MOC is responsible for: • Mission operations planning • Spacecraft operations, monitoring and control • Instrument operations execution • Mission data distribution and HKTM archiving.
5	For an obiter scenario, the science data will be received, processed and stored in the Science Operations Centre (SOC). There is no processing of science Telemetry required at ESOC. For a probe scenario the SOC will be co-located with MOC to optimise operations execution.
6	It is assumed that the SOC as part of the overall science ground segment is responsible for: • Coordinating the Science planning • Instrument command requests compilation • Providing guidelines to the instrument teams with respect to Science data processing • Archiving and long term storage of instrument data and spacecraft auxiliary data.
7	The Ground Station baseline assumes ESTRACK Ground Station network is used for the communications with the satellite. In the scenario where DSN Ground Station support is provided, operational interfaces to NASA DSN network for Ice Giants will be established.
8	X/X Ka frequencies. Communications with ground shall use X-band for TT&C at 7.2 GHz uplink, 8.4 GHz downlink and Ka Band for Science data downlink at 32GHz.
9	X Band Ranging and Doppler used for Orbit Determination. Ka doppler used during routine operational phase. Delta DOR capability used for planetary arrival. Use of navigation cameras for optical navigation.
10	All data required for the assessment of the correct functioning and performance of the spacecraft and payload are included in housekeeping data available to ESOC.
11	During interplanetary transfer, pre-separation from the Stack, all operations/ checkouts are executed via the NASA provided communications link routed through the NASA orbiter.

Assumptions	
	Options for direct communications to the ESA orbiter in the stacked configuration using the Ice Giants comms is considered.
12	After separation from the stack, ESTRACK 35m ground station passes will be routinely scheduled for communications to support nominal spacecraft monitoring control and science data dumps with the Orbiter during the commissioning, interplanetary check outs and operational phases. During critical interplanetary phases multiple ground stations will be scheduled to ensure back-up availability.
13	Routine operations are strictly preplanned using automated on-board timelines.
14	The Ice Giants spacecraft TT&C services are compliant with the applicable and, appropriately tailored standards: ECSS (PUS-C and Operability Standard), CCSDS Standards RD[83] to RD[87]
15	The OBC Mass Memory shall be compliant to PUS-C and support file based operations, storing data in separate files as specified in RD[83].
16	The exchange of files between space and ground shall follow the CCSDS File Delivery Protocol (CFDP) as specified in RD[84].

36.2.2 Ground Segment and Operational Characteristics

With respect to the ground segment and operations the Ice Giants mission has the following major characteristics impacting ground segment and operations:

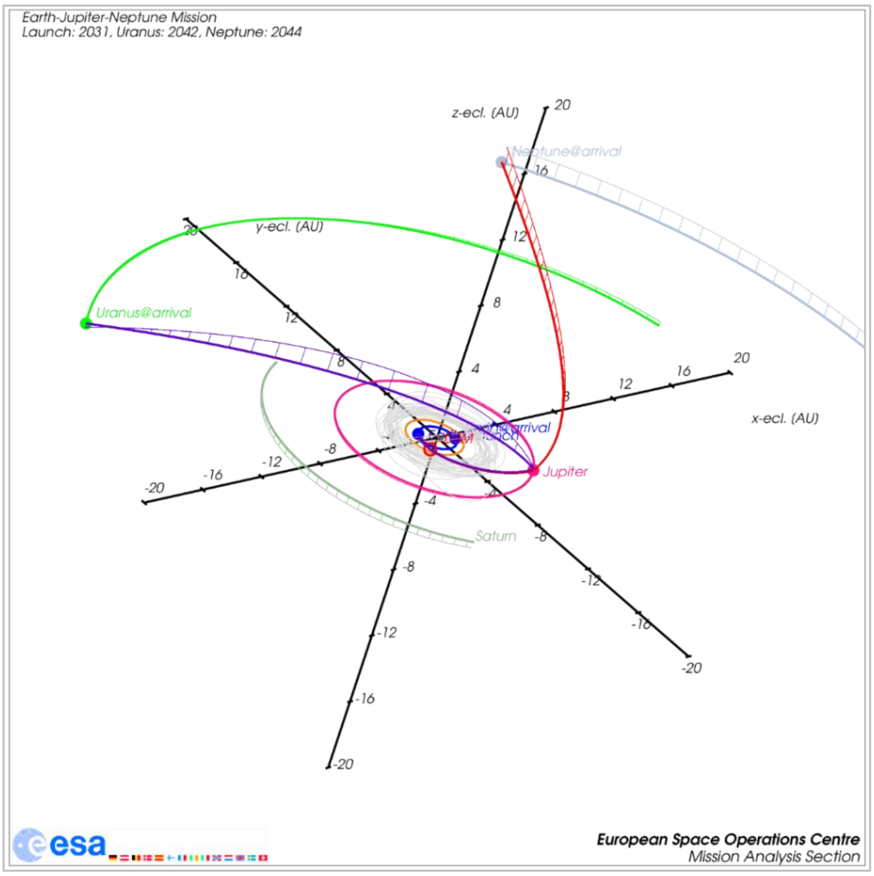


Figure 36-1: Mission Scenario

36.2.2.1 NASA Operational activity

The ESA contribution, Orbiter or Probe, will be launched under NASA responsibility with the Orbiter in a dual launch stack configuration or the Probe attached as a passenger to the NASA Orbiter.

Shortly after launch, the mission begins its checkout processes and deployment of all flight systems. During the first four weeks of cruise, continuous DSN coverage is provided for thorough characterisation of all flight systems and for accommodating the variable commanding schedules typical of early checkout operations. Once checkout is complete, the post-launch phase configures the spacecraft to low thrust SEP navigation and the DSN coverage is reduced to only 1 pass per week.

ESA Orbiter

The dual stack configuration is maintained up to Jupiter approach where the ESA Orbiter continues to be supported and under trajectory control by NASA. Prior to reaching Jupiter the ESA Orbiter prepares for operations and separation from the stack. This critical phase of the mission will be supported by both daily DSN and ESA ESTRACK coverage for commanding and tracking for 2 weeks approaching the event. Continuous ground station and FCT coverage is required during the days surrounding the separation and subsequent Ice Giants orbiter Jupiter gravity assist and manoeuvring. Independent operations then commence with ESA controlling their Ice Giants orbiter.

ESA Probe contribution

The ESA probe is operated as a passenger on the NASA stack with minimal periodic check out operations.

Starting from 9 months until Probe Entry, DSN coverage increases to daily tracks in preparation for the Probe release. During the approach phase and probe entry the HGA is continuously Earth-pointed to provide flexibility in DSN scheduling to ensure maximum science data return during this period.

36.2.2.2 Orbiter Operational considerations

Mission Timeline: Neptune Orbiter	
Activity	**Duration**
Dual Launch SLS	Depart Earth 13/2/2031
Interplanetary transfer	~ 1 year 10 months
Fly by phase (after separation from stack)	Fly by Jupiter 24/12/2032 @ 857000km
Cruise phase	~11.5 years
Orbit insertion NOI	Arrive Neptune 1/9/2044
Science Phase	2 years. Orbital period ~180 days. Target orbit period of 50days.
Disposal Phase	To be defined

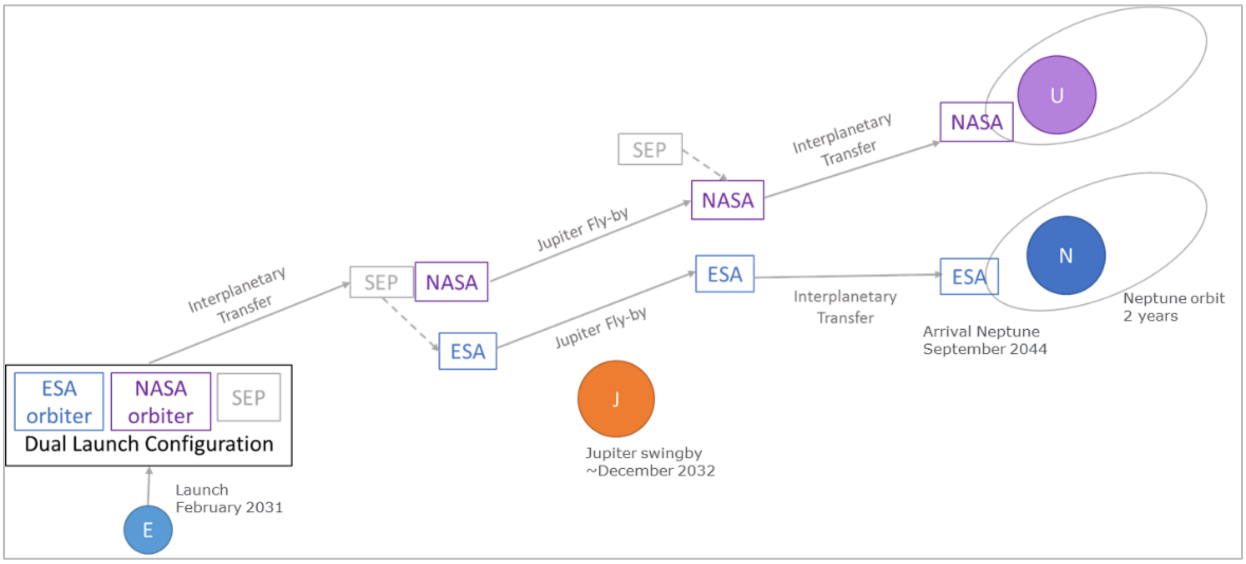


Figure 36-2: Neptune Orbiter Mission Profile

Mission Timeline: Uranus Orbiter	
Activity	**Duration**
Dual Launch SLS	Depart Earth 13/2/2031
Interplanetary transfer	~ 1 year 10 months
Fly by phase (after separation from stack)	Fly by Jupiter 24/12/2032 @ 10000km
Cruise phase	~9 years 4 months
Orbit insertion UOI	Arrive Uranus 6/4/2042
Science Phase	**2 years** Orbit period of 104.5 Earth days. Target orbit period of 37.8 days. **4 years** Orbit period of 150 Earth days. Target orbit period of 50 days.
Disposal Phase	To be defined

During the stacked cruise phase there is no power interface to the NASA orbiter. The Ice Giants orbiter provides all its own power. A SpaceWire interface to the NASA orbiter allows for the TM/TC data interface to Earth. This interface supports all TM/TC traffic covering post launch checkout requirements and the periodic health checks while in the stacked configuration. Outside of these check out periods the ESA orbiter is in a low power, sleep mode.

Post stack separation, X-Band is used for TT&C and Ka band used for the Science downlink. Data rates achievable are 42kbps at Neptune for the 0.48 Gb/day of science and 94 kbps at Uranus for the 1.09 Gb/day of science. Usage is made of 3m HGA for X and Ka band and two LGAs for X-Band.

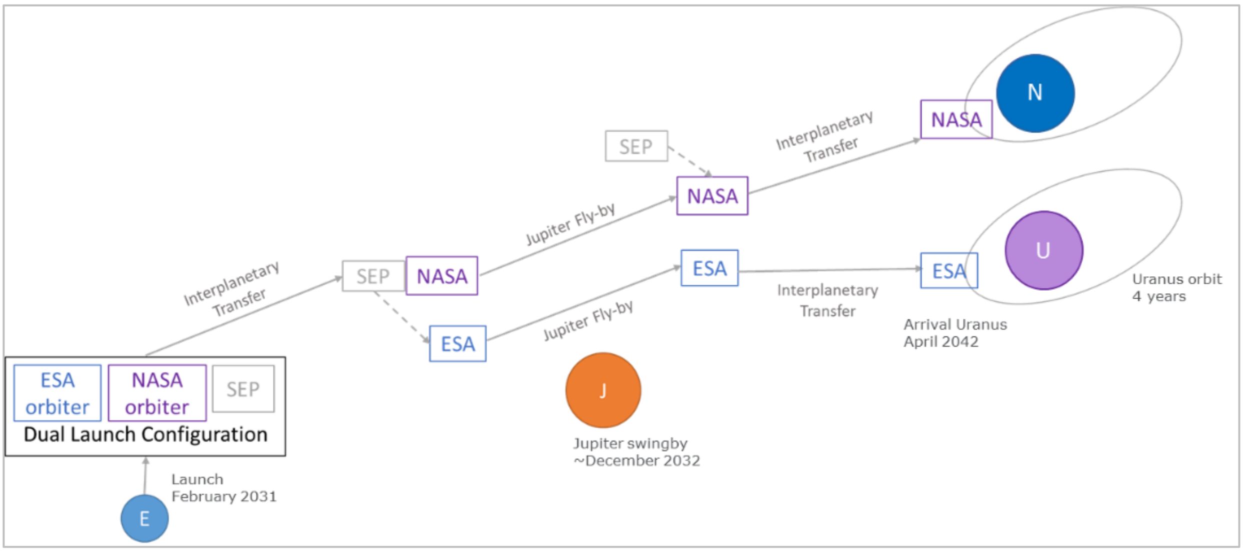


Figure 36-3: Uranus Orbiter Mission Profile

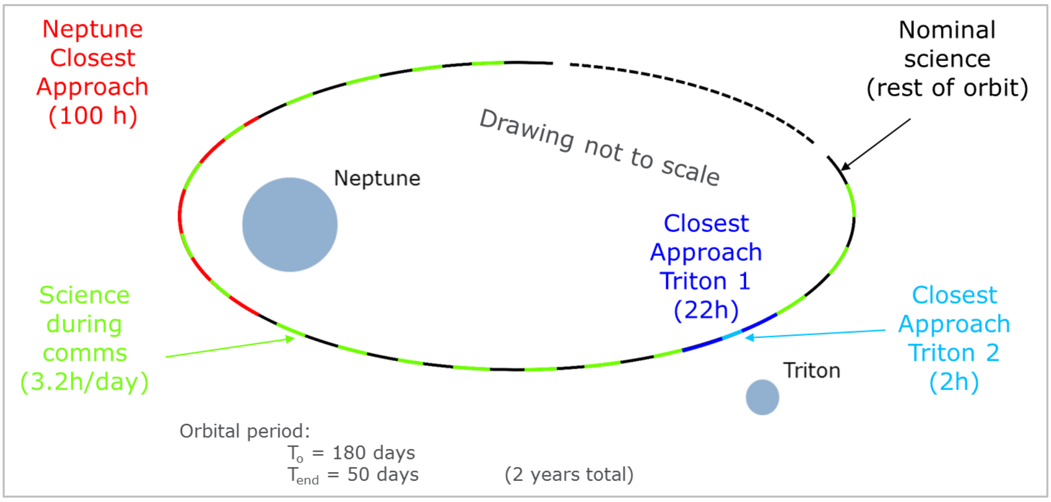


Figure 36-4: Neptune Science orbit timeline

36.2.2.3 Probe

Mission Timeline: Neptune/Uranus Probe	
Activity	**Duration**
Launch	Depart Earth 13/2/2031
Transfer	11.5 years for Uranus 13.5 years for Neptune
Release	Deploy and checkout
Transfer to Entry interface point EIP	20 days
Entry	Initial altitude 600km

Mission Timeline: Neptune/Uranus Probe	
Activity	**Duration**
Science Phase	90 minutes
Disposal Phase	Final pressure: 10 bar.

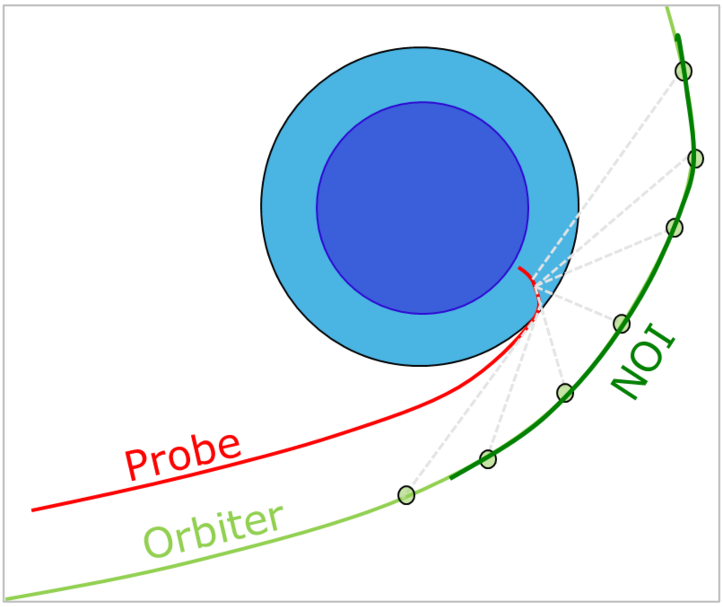


Figure 36-5: Probe Orbit insertion

Upon Probe release the Probe uses its own power system and timer switches to activate the pre-planned automatic sequences. No telecommanding capability is assumed after release. UHF is assumed for communications with minimum TM rate of 2kbps. The maximum distance of probe to relay orbiter is 40000km.

36.3 Rosetta Lessons Learned applicable to Ice Giants operations

Below are extracts from the Rosetta mission lessons learned which are considered applicable to the Ground segment and operations approach for an Ice Giants type mission. For the full set of Rosetta lessons learned see RD[91].

36.3.1 Ground Segment Incremental Development Approach

Incremental development of operations concepts and ground segment over the long cruise phases.

For missions with long cruise phases, as is the case for the Ice Giants mission, the following points should be applied to the development approach of the Ice Giant ground segment.

- Plan for an incremental development of operations concepts and ground segment; opportunity to develop operational concepts and ground systems once knowledge of the "mission system" is accrued in flight. MOC readiness for launch. Deferred SOC development.

- Plan for proficiency training of new teams on already developed elements and allow this to become opportunities to improve the system.
- Manage obsolescence of systems with care; thorough trade-offs, only if really beneficial and unavoidable.
- Actively maintain knowledge and awareness of major mission limitations, risks, and unknowns such that expectations are properly managed.

36.3.2 Team Evolution

Adopt an active strategy for planning of team evolution and knowledge preservation to cover long cruise phase.

Missions with extremely long duration from launch to the main operations phase pose a significant challenge in terms of knowledge and team evolution planning. Both aspects can be tackled with a pro-active approach involving recruitment of team members with long availability, who are then prepared for additional roles applicable to later mission phases. This addresses the preservation of knowledge and build-up of the expertise required for such a long term mission.

A similar approach should be adopted for the Ice Giants mission applying this active strategy for team evolution planning in the recruitment process and career evolution planning.

36.3.3 Operations Planning for Long Cruise Phases

Operations planning for long cruise phases to consider adopting Active and Passive cruise phases and associated related activity levels.

- Non-contact periods for "passive" cruise should be in the order of 7-14 days; anything above/below is likely to cause major impacts on the spacecraft or the ground segment design.
- Active cruise phases shall allow for unplanned activities, i.e. passive cruise phase should remain unaffected.
- During periods of low frequency contacts, passes are best placed in the middle of a working week.
- Adequate and dedicated planning tools and processes shall be put in place from the start of the mission. (e.g. planning concept, rules, commanding interfaces, operations grouping/sub scheduling, etc.)

36.3.4 Planning of Complex Mission Phases

Mission planning concepts, timelines, and tools shall be designed as simple as possible and reduced to their essential elements; simplicity is the key to guarantee the flexibility ensuring the possibility to add more elaborated features as needed.

As far as possible, flight and ground segment recurring events (e.g. manoeuvre, planning) should be synchronised with calendar cycles; this ensures an efficient utilisation of the mission control team.

36.3.5 In-Flight Characterisation

Plan for in-flight power, thermal and disturbance torques characterisation phases with appropriate expert support/tools.

In flight problems can occur with inaccurate prediction of disturbance torques, power and thermal behaviour. Planning for in flight characterisation campaigns with an adequate support of expertise and correlation of tools allows for optimisation of the inflight, power, thermal and AOCS systems,

36.3.6 Availability of Engineering Model (EM)

Availability of the Orbiter or Probe Engineering Model (EM) hardware to Flight operations teams post LEOP with adequate documentation and training mitigates the need for long Phase E2 industrial support contracts.

36.4 Baseline Design

36.4.1 Ground Segment Overview

The ground segment will be set up according to baselines established for the Solar System & Exploration Missions as performed by ESA. The Ice Giants mission will be operated by the ESA Mission Operations Centre (MOC) located at ESOC in Darmstadt, Germany. The spacecraft operations, scientific instrument operations, flight dynamics and ground station activities will be performed from the Mission Operations Centre (MOC).

The planning of science operations and co-ordination of the scientific input will be the responsibility of PI teams via a Science Operations Centre (SOC) forming part of the Science Ground Segment (SGS). For the Probe, the SOC would be co-located with the MOC, for the Orbiter mission the SOC would be located at ESAC, near Madrid, Spain.

36.4.2 Ground Segment Development Approach

36.4.2.1 Orbiter

Reviewing lessons learned from Rosetta, as well as the approach taken for the JUICE mission ground segment development and operations, a staggered development approach would be established for the Ice Giants ground segment.

The MOC and SOC ground segment requirements would be established pre-launch. The MOC Ground Segment is required to support Mission Operations from Launch while the SOC Ground Segment has a deferred development due to the long interplanetary transfer periods.

MOC development would be split into a “Launch system”, linked to launch date, and would be based on Launch, LEOP and transfer operational requirements. This would cover all activities up until shortly before arrival at the target planet. The “Planetary System” MOC development would be linked to planetary orbit insertion date and based on the operational requirements to support planetary phase of mission, orbit insertion, maintenance and science operations execution.

The MOC “Planetary System” would be considered as an iteration of the ‘Launch System’ MOC ground segment. At the planetary arrival stage of the mission the operations concepts would have matured with in-flight experience and the subsequent development of the "Planetary System" benefits from the improved mission system knowledge.

The SOC system is developed to support science data processing, science mission planning and operations in the target planet orbit. The development schedule for the SOC is linked to planetary orbit insertion dates.

Example "Launch System" Ground Segment Development		
Review name	Relative to Launch	System participants
Ground Segment Requirements Review	L-4.5years	MOC and SGS
Ground Segment Design Review	L-3.5y	MOC
Ground Segment Implementation Review	L-1.5y	MOC
Ground Segment Readiness Review feeding into Flight Readiness Review	L-4months	MOC
Operations Readiness Meeting	L-1m	MOC
Example "Planetary System" Ground Segment Development		
Review name	Relative to Orbit Insertion	System participants
Planet Operations Design Review	OI-3years	MOC and SGS
Planet Operations Implementation Review	OI -20months	MOC and SGS
Planet Operations Readiness Review	OI -8m	MOC and SGS
Planet Orbit Insertion Readiness Review	OI -1m	MOC
Moon Orbit Insertion Readiness Review	Moon OI -1m	MOC

Table 36-1: Example of Ground segment development timelines.

36.4.2.2 Probe

For the ground segment development of a Probe type mission the approach is simplified. For the MOC, a small MOC is developed to be ready at Launch to allow for pre-launch Probe checkout and testing. There would be minimal activity during the planetary transfer phase with occasional check out and testing of the Probe.

For the SOC, a small SOC co-located with MOC would be developed to allow for joint mission execution and science timelining. A delayed start of development of the SOC linked to planetary insertion would be adopted due to the long cruise phase.

36.4.3 Mission Operations

36.4.3.1 Orbiter

The MOC is heavily involved pre-launch for the development and support to Orbiter system validation testing, integration testing, readiness and preparation of MOC systems and Flight control teams to support launch. These MOC activities start '~Launch – 5.5 years'.

During the Launch and LEOP the MOC executes pre-launch final checks on the orbiter and executes check out of subsystems performance post launch. The full launch systems Full Flight Control Team (FCT) is in place for these activities.

For the periods of interplanetary transfer as the passenger in the combined stack (~1 year 10 months), orbiter separation before reaching Jupiter and Jupiter flyby, there are periods of intense flight operations activities i.e. orbiter commissioning, stack separation, Jupiter flyby, interspersed with quieter transfer periods where only periodic health checks are executed. To support these activities varying levels of FCT support are scheduled over these periods.

Post Jupiter flyby during the ~11 year cruise phase to Neptune (9.3 years to Uranus), where only periodic health checks are carried out, minimal FCT support is planned for.

Planetary arrival operations preparation activities start at approximately 'Orbit insertion-4 years'. The FCT is ramped up over this period as the MOC is developed for planetary phase of the mission and the science operations system are developed and finalised. The operational experience and orbiter familiarisation gathered during cruise phase is used to optimise the operations concepts used in the planetary phase.

36.4.3.2 Probe

Pre-launch activities in the MOC take place to support Probe system validation testing, integration testing, readiness and preparation for launch.

Probe mission operations during the cruise phase are minimal. Approximately six months prior to Probe release a period of intense preparation and validation activities is executed by both MOC and SOC collocated at the MOC. Final detailed mission preparation and system validation, takes place during Probe cruise period (20 days) post release from the orbiter.

Full operational support is provided from both MOC and SOC teams during Probe entry and descent period noting that the one way light time delay of 2.7 hrs for Uranus, and 4.3 hrs for Neptune is such that the Probe mission is completed by the time the TM data reaches Earth.

36.5 Ground Stations

The three ESTRACK 35m ground station sites Malargüe, New Norcia (NNO-1), and Cebreros are baselined to support the operational needs of the Ice Giants mission in X / X Ka. The upgraded cryogenic capability has been assumed to be available in all ground stations supporting Ice Giants in both X and Ka band feeds. This cryogenic capability improves the G/T assumed to be available at the different ground stations and has been used in the TT&C link budget calculations.

The Ka-band uplink frequency will be allocated to the Radio Science Ka-Band transponder with an uplink frequency in the 34200 – 34700 MHz frequency band. For the Radio Science experiments both X and Ka band uplinks would be available in parallel during nominal operations in orbit.

Varying ESA 35m ground station support requirements are evident during the early stages of the mission dependent on the planned activities. During stacked transfer phase all comms is via NASA orbiter capability. A possible ESA ground station contribution to Launch and LEOP could be considered during this stacked phase.

During routine science phases (~2 years) daily ground station passes are expected of ~ 3 to 4 hours per day. The data downlink duration is constrained by on board power availability rather than ground station visibility. The 35m ground station visibility snapshot (Neptune, Uranus from Earth) is shown in Figure 36-6. It can be seen from this plot that the ~ 3 to 4 hours coverage from a single ground station is easily met. NASA DSN contributions to ground station time can also be considered.

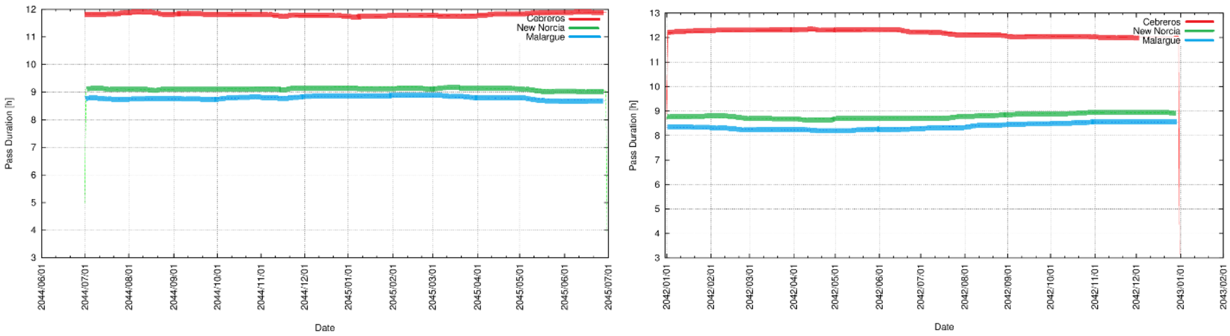


Figure 36-6: 35m Ground station visibility to Neptune(l) and Uranus (r)

36.5.1 Seasonal Solar Conjunctions

Solar conjunctions occur at both Neptune (every May for ~7 to 8 days) and Uranus (period of ~8 days in August shifting yearly) resulting in RF interruption. During these solar conjunction periods, i.e. within <3 deg of the Solar conjunction, no flight operations shall take place and the spacecraft shall be put into a safe configuration.

An additional operational constraint is that NO critical operations e.g. flyby, deep space manoeuvres, orbit insertion etc. occur within 5 deg of the Solar conjunctions. Activities need to be planned such that these conjunction periods are avoided.

36.5.2 Use of UHF Telescopes

It is proposed for support to the Probe mission that a large array (20) of geographically distributed Radio Telescopes is deployed on ground during the probe descent mission to deduce additional flight dynamics information. This would be done by locking onto the Probe carrier signal. The UHF Radio telescope arrays provide Probe 'alive' signal detection.

36.5.3 Enhanced Ground Station: Arrayed Antennas

Studies have been carried out to investigate the feasibility of improving the 35m ground station performance by the arraying of 35m antennas. Two studies have been carried out through OPS-GS. (RD[88], RD[89], RD[90])

The results of the studies show that the use of two antennas would give ~3dB improvement on data rate and use of further antennas would further increase the data rate performance.

With Ka-Band there are constraints on the proximity between the terminals used to array, however these constraints could be respected in Malargüe where ESA has full control of the site.

Two possible implementations could be supported;

1. At MLG: Implement one additional 35m terminal in Malargüe, MLG-2, Rx only in X-Band and Ka-Band (with optional X-Band uplink), with the same performance as MLG-1. MLG-1 and MLG-2 can then be arrayed to support the Ka-Band and X-Band downlink.
2. At CEB:
   1) Implement one additional 35m terminal in Cebreros, CEB-2, Rx only in X-Band and Ka-Band (with optional X-Band uplink), with the same performance as CEB-1.
   2) Establish a cooperation with NASA/JPL in Robledo, to array CEB-1 with one of the NASA 34m BWG antennas, which have similar performance as the cryo-feed enhanced CEB-1. Such arraying of ESA/NASA antennas would utilise the proposed NASA DSN contribution. If there is an issue linked to distance between NASA Robledo and CEB terminals, NASA/JPL could be requested to support from Robledo with a full NASA/JPL array configuration.

It is recommend to further analyse the application of arraying of large antennas (ESA and/or NASA DSN) for use by the Ice Giants mission.

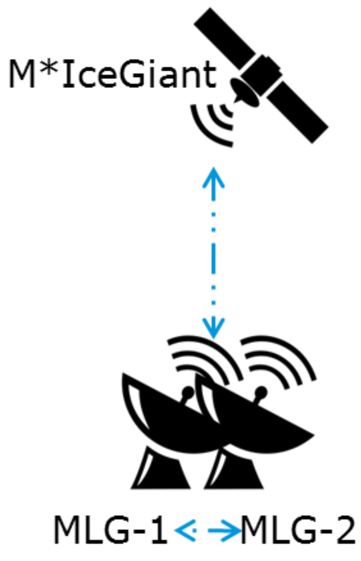


Figure 36-7: Arrayed Malargüe two antenna example

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37 RISK ASSESSMENT

37.1 Reliability and Fault Management Requirements

The following reliability and fault management requirements were proposed for the IceGiants mission.

ID	Requirement
REQ-010	The overall reliability of the mission shall be ≥ 85% at end of life (loss of S/C).
REQ-020	The lifetime* of S/C shall be compatible with the mission requirement.
REQ-030	Single-point failures with a severity of catastrophic or critical (as defined in ECSS-Q-ST-30C/40C) shall be eliminated or prevented by design.
REQ-040	Single-point failures (other than catastrophic or critical) shall be avoided in the design of the mission units.
REQ-050	Retention of single-point failures of any severity rating in the design shall be declared with rationale and is subject to formal approval by ESA. Multiple failures, which result from common-cause or common-mode failure mechanisms, shall be analysed as single failures for determining failure tolerance.
REQ-060	A failure of one component (unit level) shall not cause failure of, or damage to, another component or subsystem within and between mission units.
REQ-070	The failure of an instrument shall not lead to a safe mode of the mission units.
REQ-080	The design shall allow the identification of on-board failures and their recovery by autonomously switching to a redundant functional path. Where this can be accomplished without risk to spacecraft and instrument safety, such switching shall enable the continuity of the mission timeline and performance.
REQ-090	Where redundancy is employed, the design shall allow operation and verification of the redundant item/function, independent of nominal use.
REQ-100	The design and its operation of shall be compliant with applicable Space Debris rule** in all phases of its lifecycle. *(e.g. ESA/ADMIN/IPOL Space Debris Mitigation for Agency Projects)*
REQ-110	The S/C design and its operation shall be compliant with applicable 'planetary protection'-requirements in all phases of its lifecycle. *(e.g. ESSB-ST-U-001)* *Remark: requirement is applicable only for the option: lander/ Neptune moon Triton*
REQ-120	The S/C design and its operation shall be compliant with applicable launch requirements in all phases of its lifecycle. *(e.g. CSG Safety Regulations)*

Overall remark: *because this study was dedicated to identify technological developments many of the requirements ('gray' marked) are used for orientation rather than for the full implementation in the study baseline and has to be verified during a follow up study (see chap. 37.6)*

see applicable mission criteria's* **Table 37-2

*** depending on the responsible launch authority and/ or launch operator*

Table 37-1: Reliability and Fault Management Requirements

The requirements were reviewed during the course of the study and found to be adequate for the IceGiants CDF-Study.

37.2 Risk Management Process and Scope of Risk Assessment

Risk management is an organised, systematic decision making process that efficiently identifies, analyses, plans, tracks, controls, communicates, and documents risk in order to increase the likelihood of achieving the project/ study goals. The procedure comprises four fundamental steps:

- **Step 1:** Definition of the risk management policy which includes the project success criteria, the severity & likelihood categorisations, and the actions to be taken on risks
- **Step 2:** Identification and assessment of risks in terms of likelihood and severity
- **Step 3:** Decision and action (risk acceptance or implementation of mitigating actions for the risk reduction)
- **Step 4:** Monitoring, Communication and documentation and risk acceptance.

The Ice Giants CDF-Study is a pre-phase A feasibility assessment. It focused on the identification of technological developments necessary to make this mission feasible. Therefore the focus of the Risk assessment was on the first two steps whereby solely the severity of unwanted consequences were judged in step 2. The further steps are mentioned in this chapter for the completeness in terms of the overall Risk management process.

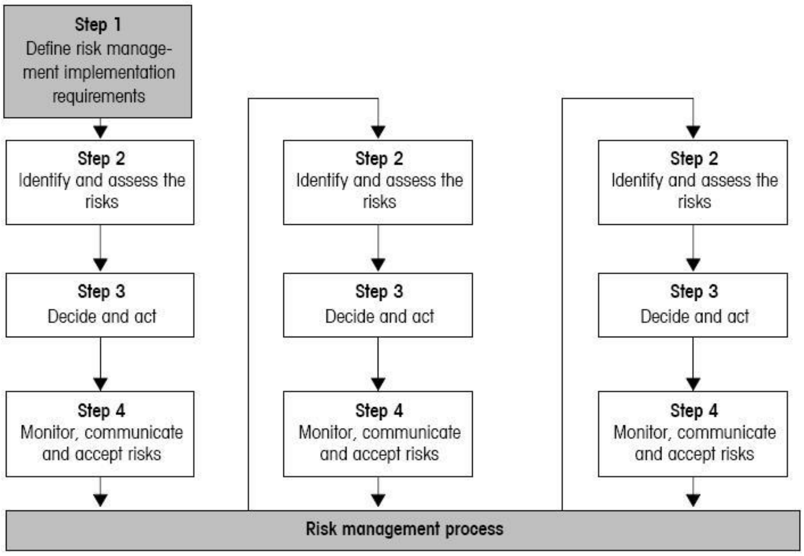

Figure 37-1: ECSS-M-ST-80C, 2008 Risk Management Process

The results of the 2 dedicated steps have to be seen as preliminary. The full documentation of the Risk assessment is pre-mature.

The basis for the preliminary risk assessment is the kick-off documentation/ presentation of the study. Changes in the kick-off baseline which are caused by identified risks were already seen as mitigation measures.

The scope of the preliminary risk assessment was clearly defined at the beginning and during the study. The risk assessment comprises all mission phases and mission elements whereby the focus was on the Neptune/ Uranus-probe followed by the

Neptune/ Uranus-Orbiter. Beside the probe and the orbiter also a lander was discussed during one session of the study resulting in a very preliminary risk portfolio.

The preliminary risk assessment for Ice Giants study is considering risks for the following mission elements (options to contribute to a NASA (opportunity) mission):

- Option 1.1/ 1.2: a Neptune/ Uranus probe or ..
- Option 2.1/2.2: a Neptune/ Uranus orbiter or ..
- Option 3: a lander (Neptune moon Triton).

Furthermore the risk during the following pre-project, project and mission phases were identified:

- Study (design maturity in pre-project phase)
- Development phase (technological maturity in pre-project phase)
- Mission realisation (project phase)
- Launch (project/ mission phase)*
- Interplanetary trajectory to mission target including orbit insertion (Neptune/ Uranus)
- Descent (probe and lander) and landing phase (for lander only)
- Exploration phase (orbiter and probe)

**the launch was considered in terms of the assumed study baseline launcher SLS.*

37.2.1 Approach for Risk Identification and Risk Reduction (steps 2 and 3)

The assessment of the specific risks presented in chap. 37.5 based on the overall approach for the hazard description [Figure 37-2] visualised hereafter.

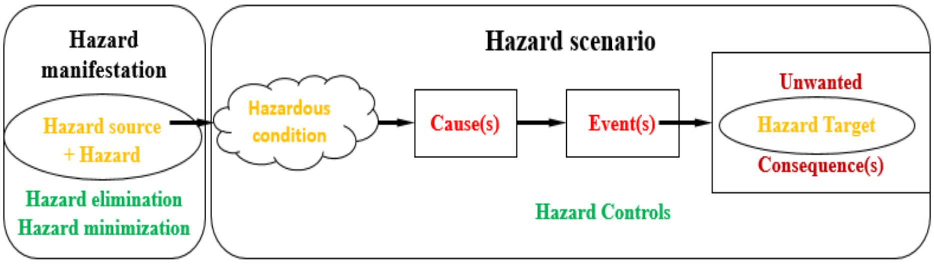

Figure 37-2: Risk identification and risk reduction

The assessment started with the definition of the 'Hazard Source', the 'Hazard' and the 'Hazard Target'.

In the next step the 'primary Hazardous Condition' which is inherently connected to the Hazard Source, the Hazard and the Hazard Target will be identified including the expected 'Unwanted Consequences'.

Finally the 'Cause' (e.g. the failure modes) which is triggering the 'Event' originating the Unwanted Consequence will be recognised. The occurrence of the Cause, its transition to an Event (or Event Chain) and the realisation of the Unwanted Consequence is often influenced by circumstances summarised as 'secondary Hazardous Conditions'.

Based on this information the likelihood of the occurrence of the Unwanted Consequences can be judged as point estimate which applies in general to the 'worst case' Severity category.

In case the risk is not acceptable in terms of the used Risk Index (see chap. 37.3.3) Risk Reductions via Mitigation Measures has to be defined to bring the risk in an acceptable area of the Risk ranking. Such Mitigation measures like Hazard Elimination, Hazard Minimization and Hazard Controls are beyond the baseline. They have to be considered in a delta study or in the project/ mission phase.

An Initial Risk for one Hazard Target can be connected or lead to a new(^)/ additional(+) risks for another Hazard Targets as a consequence of its reduction e.g. the mitigation of Dependability risks (e.g. increase of the redundancy) can lead to an impact on other Hazard Targets like programmatic (e.g. possible overrun of the mass budget) and/or cost and/ or schedule. Such risk propagation is visualised in Figure 37-3 hereafter.

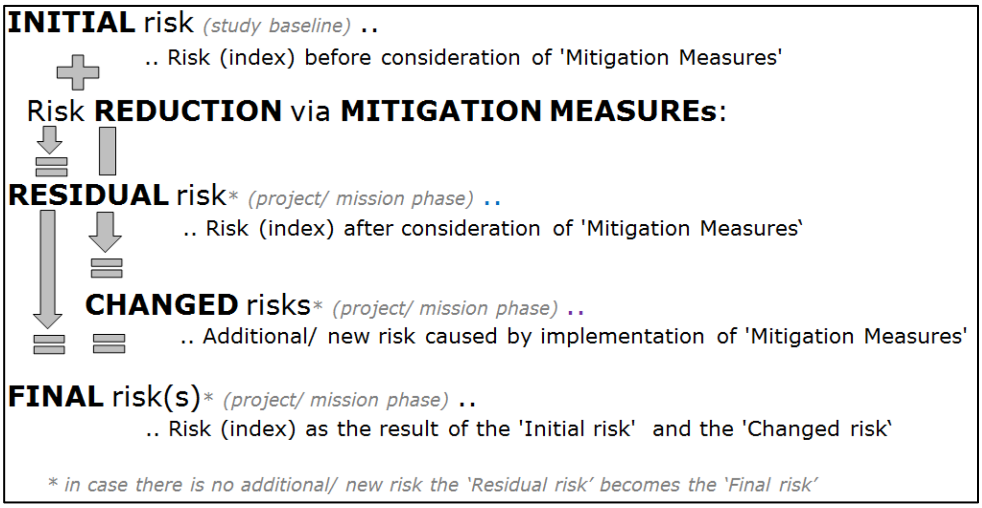

Figure 37-3: Risk propagation

The terms used in connection with the risk identification are defined as:

Hazard[1] [H]	Existing or potential property/ state[2] of a Hazard Source that can result in a mishap for the Hazard Target
Hazard Source [HS][3]	An item/ entity of the CDF study and/ or space mission
Hazard Target [HT]	An item/ entity/ person which could get affected by the mishap like performance (science, services, ..)/safety (harm, damage)/ cost/ schedule *(see chap. 37.3.1)*
Hazardous Condition[2] [HC]	Hazardous conditions are levels/ borders capacities or situation/ circumstances which can initiate a 'Hazard scenario' and can be associated . * . with physical, chemical or biological capacity of a 'Hazard source' (primary condition [HCp]) – intrinsic property but also with the technology, design, manufacturing, organisation (secondary conditions [HCs]) – functional/ physical state
Hazard Manifestation	The Hazard Source with its potential Hazards and Hazardous

	Conditions becomes part of the study baseline/ future mission
Hazard Scenario	The combination of 'Cause(s)' and 'Event(s)', which results into a specific Unwanted Consequence
Cause [C]	Root Cause which is the origin of a Hazard Scenario
(final) Event [E]	Final physically event or status which is directly leading to the Unwanted Consequence under the given Hazardous Conditions
Event Chain	Between the Cause and the Event several intermediate events might occur
Unwanted Condition [UC]	Is a/are potential result(s) of a Hazard Scenario which specified the negative effect for the Hazard TARGET[HT] in the frame of the CDF study the Unwanted Conditions has to be specified based on the Study/ Mission Success Criteria's (see chap. 37.3.2)
Hazard Elimination[4]	The Hazard will be fully eliminated mostly by elimination of the Hazard Source
Hazard Minimisation[4]	The Unwanted Consequences (Severity category) will be downgraded mostly via changes in the primary Hazardous Condition
Hazard Control[4, 5]	Engineering or administrate measurements

Remarks:

1/3 *Hazards are NOT events (neither accidents nor incidents) but potential threats to the Hazard Target;*

2 *Property or state which can be associated with the design, manufacturing, operation, organisation, application or environment,*
an intrinsic property of an item/ entity, e.g. instable isotopes/ radiation, Hardware/ sharp edges,
- a functional/physical state of an item/ entity e.g. Medium/ high pressure in a vessel; Hardware/ high temperature of a surface, ...

3 *Prerequisite(s) for the occurrence of 'Hazard scenarios' with their negative effects ('Unwanted Consequences') on 'Hazard Target(s)*

4 *Basic strategies/ Mitigation Measures for the Risk Reduction*

5 *e.g. - Design selection (failure tolerance, ..)*
- *Design to minimum risk (Safety margins/factors)*
- *Automatic safety device, design to contain,*
- *Warning device, crew escape/ safe haven,*
- *Dedicated procedures, regulations, standard's, programmes, ...*

However, because of the development-oriented focus of the study, risk mitigation measures were mostly discussed in terms of the identification of equipment/ material development.

37.3 Risk Management Policy

37.3.1 Hazard Targets

The CDF risk management policy for Ice Giants study aims at handling risks which may cause serious programmatic/ cost/ schedule***/ technological, performance (science return [or] services) **/ technical and protection* impact on the future project.

Because of the development-focused objectives of the study, safety aspects as defined hereafter were not assessed.

* *'Safety' related to the human life and health has a higher priority and importance than 'Safety' related to property and environment. To have a clear split between both safety aspects in the report the term*

- *'safety' is used exclusively for risks related to human life and health on ground and in space*
- *'protection' is used exclusively for risks related to equipment (e.g. the S/C, the launcher), property (e.g. launch facility) and planetary environments (terrestrial, space and specific solar objects)*

*** 'Performance' is standing for e.g. 'science' incl. 'technological tests' or 'services' (e.g. telecommunication, navigation , cargo)*

**** The Hazard Target 'Schedule' has two aspects:*

- *cost related .. each delay might lead to a project extension, shift of launch preparation and the (c+sh) launch; what is usually linked to a cost increase; but this does not mean that the mission can not be performed at a later date/ launch opportunity from launch/launcher-viewpoint*
- *mission related .. schedule-constraints like launch window or Earth-orbit escape window has to be (m+sh) considered depending from several mission conditions, like mission destination, Earth-orbit before escaping to e.g. the interplanetary trajectory and the trajectory itself.*

37.3.2 Success Criteria

The success criteria with respect to the program, science, technical, safety/ protection safety, schedule, and cost objectives are presented in Table 37-2:

Risk Domain (Hazard Targets)	**Success Criteria**
Programmatic	**STU1:** preliminary feasibility study for several mission options dedicated to the Ice Giants Neptune and Uranus **MIS1:** Contribution* to a NASA opportunity mission dedicated to observe one of the Ice Giants (Neptune+Triton/ Uranus)** *Remark:* **either an orbiter or a probe (Neptune/ Uranus) or an lander (Neptune moon Triton)* *** either Neptune and its moon Triton or Uranus*
Programmatic (Cost)	**PR-C1.a/b:** Orbiter .. CaC for ESA < 550M€ (2016 EC) ->M Class Mission .. **PR-C1.b :** Probe .. CaC for ESA < 550M€ (2016 EC) ->M Class Mission .. **PR-C1.1b :** Lander .. CaC for ESA < 550M€ (2016 EC) ->M Class Mission .. **PR-C2:** The mission design shall follow a "design-to-cost" for all its elements under ESA responsibility **PR-C3:** The mission and system design should make use as far as practicable of technologies from suppliers from ESA Member or Cooperating States. *Remark: Use of equipment subject to US export control regulations shall be agreed on a case-by case basis with the Agency.*
Programmatic (Schedule)	**PR-S1:** All architecture elements are available and their FRR successful for the launch (NLT 2029 ..34) **PR-S2:** The contributions from international partners are available at the relevant milestones of the development schedule **PR-S3:** TRL >6 for all components at the time of mission adoption (est. 2022) **PR-S4:** Low development risk during Phase B2/C/D ** ISO scale 2016* **PR-S5:** Delivery to NASA app. 16 month before launch (mid Oct. 2029 with Launch in February 2031)

Risk Domain (Hazard Targets)	**Success Criteria**
Performance (Science/ Services) **Technical**	**PER1.1a:** Orbiter .. to transport a probe to Neptune or lander to Triton and orbiting/ monitor of Neptune **PER1.2a:** Orbiter .. to transport a probe to Uranus and orbiting of/ monitor Uranus **PER1.b :** Probe .. to enter and monitor Neptune or Uranus atmosphere **PER1.1b:** Lander .. to land on Neptune moon Triton **TEC1:** The SC operates successfully over the designated mission lifetime of 4years** **TEC2:** A reliability of >85% at the end of mission/ program [CDF Study requirement]
Design	**DES1:** Baseline launcher SLS
Safety / Protection	**SAF1:** Catastrophic hazard (2 Failure/Error Tolerance), critical hazard (1 Failure/Error Tolerance) incl. undesired incl. human performance (human related error/failure) [ECSS-Q-ST-40C] **SAF2:** No SPF can lead to catastrophic hazards; No performance degradation owing to SPF, and no failure propagation. [ECSS-Q-ST-40C] **SAF3**: Mission shall be compliant with applicable 'Launch Requirements' *(e.g. CSG Safety Regulations)* [ECSS-Q-ST-40C] **PRO:** Mission shall be compliant with applicable 'Planetary Protection Policies' *(e.g. ESSB-ST-U-001)* [ESSB-ST-U-001] **PRT3:** Mission shall be compliant with ESA policy for space debris mitigation ESA/ADMIN/IPOL(2014)2 [ECSS-U-AS-10C] **PRT3 criteria not discussed during study sessions -**

Reference: Ice Giants NASA study 2017(JPL D-100520)/ / overall CDF study requirements/ ECSS-Q-ST-40C

Table 37-2: Success Criteria

37.3.3 Severity Categorisations

For the Ice Giants CDF-study a preliminary risks identification for all Hazard Targets like programmatic(pr) in terms of e.g. cost(c), schedule(sh), technological readiness (tr), performance(dp)*/ technical(dt) and safety(s)/ protection(p) was performed as described in chap. 37.2.

** 'Performance' is standing for e.g. 'science' incl. 'technological tests' or 'services' (e.g. telecommunication, navigation , cargo)*

The severity of the risk scenarios are classified (based on the study baseline) according to their Hazard Target of impact. The consequential severity category of the risks scenarios is defined according to the worst case potential effect with respect to programmatic and science / performance objectives, technical and safety/ protection objectives, schedule objectives and/or cost objectives (see Table 37-3).

In addition, identified risks that may jeopardise and/or compromise the Ice Giants mission will be ranked in terms of severity of unwanted consequence (shortened as 'severity of consequence') for the study baseline.

The scoring scheme with respect to the severity of consequence on a scale of 1 to 5 is established in Table 37-4, based on recommendations given for the risk assessment in ECSS-M-ST-80C.

Severity Score	**Dependability** **Performance**(Science return - dp) & **Technical** (Dependability – dt)	**Safety & Protection** (s/p)	**Schedule** (pr/ sh) incl. technological readiness (pr/ tr)	**Cost** (pr/ c)
Catastro-phic 5	Performance: * Failure leading to the impossibility of fulfilling the mission's performance Technical: failure propagation: * from lower system level to highest system level * from mission to constellation/ campaign level * leading to loss of safety-related barriers	Safety: * Loss of life, life-threatening or permanently disabling injury or occupational illness; * Loss of an interfacing manned flight system Protection: * Loss of the system (e.g. S/C) * Severe detrimental environmental effects * Loss of launch site facilities.	Delay results in project cancellation .... For the project the 'Schedule' is also mission related because of the defined launch window (..??.. weeks) and escape windows (..??.. week) defined by the mission destiny	Cost increase result in project cancellation
Critical 4	Performance: * Failure resulting in a major reduction (70-90%) in overall performance according mission objective Technical: * Major damage to flight systems	Safety: * Temporarily disabling but not life- threatening injury, or temporary occupational illness; Protection: * Major detrimental environmental effects. * Major damage to or ground facilities. * Major damage to public or private property	Critical launch delay (24-48 months)	Critical increase in estimated cost (20 -50%)
Major 3	Performance: * Failure resulting in a major reduction (30-70%) in overall performance Technical: * Major degradation of the flight system	Safety: * Minor injury, minor disability, minor occupational illness. Protection: * Minor system or environmental damage	Major launch delay (6-24 months)	Major increase in estimated cost (10 -20%)
Significant 2	Performance: * Failure resulting in a substantial reduction (10-30%) in overall performance Technical: * Minor degradation of system (e.g.: system is still able to control the consequences)	Safety: * Impact less than consequences defined for severity level '3- Major'	Significant launch delay (3-6 months)	Significant increase in estimated cost (5 – 10%)
Minimum 1	Performance: * No/minimal consequences (0 - 10%) in overall performance Technical: * No/ minimal consequences	Safety: * No/ minimal consequences * Space Debris Mitigation: casualty risk <10E-4	No/ minimal consequences (1-3 month delay)	No/ minimal consequences (<5%)
No 0	Initial risk fully eliminated	Initial risk fully eliminated	Initial risk fully eliminated	Initial risk fully eliminated

* *'mission' stands for a '.. set of tasks, duties ..' ECSS-S-ST-00-01C; para. 2.3.139*
** *'system' stands for a '..set of interrelated or interacting functions constituted to achieve a specified (mission) objective..' ECSS-S-ST-00-01C; para. 2.3.212*

Table 37-3: Severity Categorisation

37.3.4 Risk Acceptance Policy

The risk index is usually the combination of the likelihood of occurrence of the consequences and the severity of consequences of a given risk item.

However, in the frame of the Ice Giants study the use/ categorisation of the 'likelihood' would be only relevant in terms of 'safety' and 'protection'. Nevertheless, only the Hazard Targets 'protection' in terms of the loss of the S/C was seen as relevant for the defined study objectives (*see REQ-010 in* Table 37-1). Its rough quantitative consideration based on expert judgement.

The Risk rating of low risk (green), medium risk (yellow), high risk (red), and very high risk (dark red) were assigned based on the criteria 'Severity' (*see* Table 37-4).

37.4 Risk Drivers

The following risk drivers have been considered in the identification of specific risk items:

- New technologies (TRL)
- Design challenges (configuration, mass, volume, power, lifetime, mission/ performance operation, communication, ...)
- Major mission events (launch/ orbit insertion)
- Functional and dependability issues (performance + technical, reliability in terms of loss of S/C (e.g. single point failures – SPFs)
- Safety, and Environmental & Property factors (protection)
- Programmatic factors (cost budget, project delays).

The study approach has to answer in the first row the question whether or not the technical basis for the expected mission objectives might be available at the foreseen project start. This means: is it feasible a space system that 'WORKs' – will it satisfy the expected performance in terms of science results or space born services (quality and amount)?.

In an iterative way the design had to be checked e.g. in a follow-up study whether (or not) the space system 'WORKs GOOD'. This means, will it satisfy the expected programmatic constraints (e.g. cost, schedule) and performance in terms of safety (reasonable low numbers of safety events) and reliability (loss of S/C and, number and circumstances of anomalies).

Both, the performance and its safety/ reliability/ availability are from viewpoint of the study and mission objective inherently linked to each other and can not be fully separated from each other in terms of its Risk Assessment. Therefore a follow-up study is anyway needed.

37.5 Top Risk Log (preliminary)

Top risk items have been preliminary identified at the mission (ESA) levels. Please refer to Table 37-4 for a complete list of preliminary identified top risks and their corresponding suggested mitigating actions.

The Risk Index results reflecting the initial risk assessment* are summarised in Table 37-4. Table 37-4 summarises the final assessment considering mitigation measures as described in the Table 37-5 to Table 37-7.

The risks are sorted and marked* according the study/ mission timeline**:

- Study
- Mission Design + realisation
- Launch (preparation) + LEOP & IOT (S/C deployment)
- Cruise + Mission deployment
- Mission performance
- Other risks (e.g. interfacing risk with the mission partners like other Space agencies)
- Overall Cost (OC) + Overall Schedule (OS).

** the underlined abbreviations are used in the risk tables as the beginning initials of the Risk no.*

*** appearance of '(root) cause' and 'events' (chap. 37.2.1) in the study/ mission timeline*

The risk numbering (1st column of risk tables) is associated to the study internal risk allocation and does not give a ranking according their importance or any other numerical order.

IMPORTANT:

Safety/ protection and reliability related risks were not in the focus of this study. However, this does not mean that these risks are from subordinate importance. It is because safety, reliability is often effected by and interacting with several other risks scenarios via randomised & systematically failure/ errors (lifetime, launch/space / operation environment, TRL, design fluffs, ...) or contribution to overall mass, power budget, cost and schedule risks (e.g. via redundancies, safety factors and margins, S/C modes, implementation and verification of related mitigation measures...).

The risk assessment in terms of the safety and reliability has to be part of the follow-up studies.

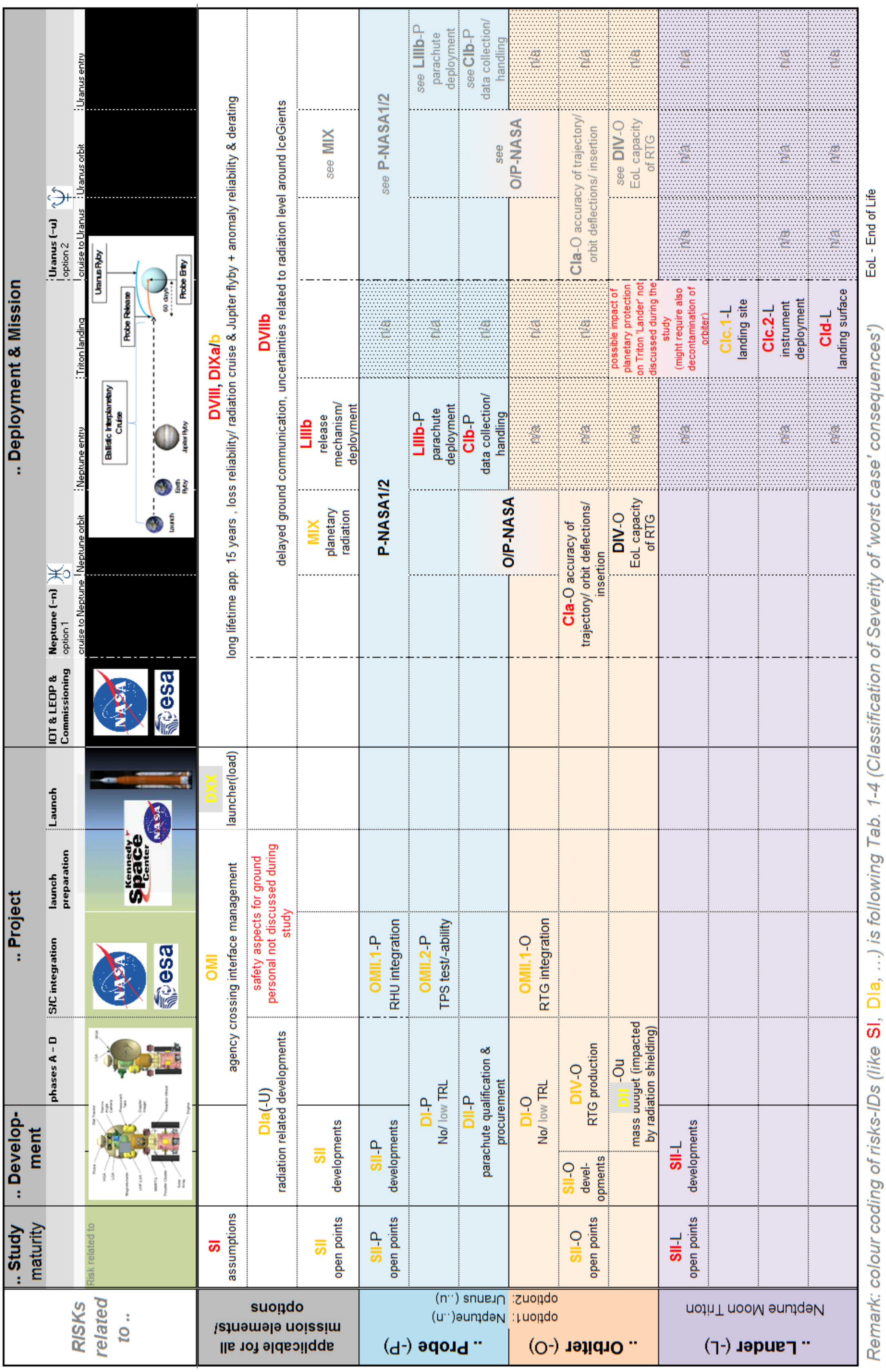

Figure 37-4: Risk Log summary

The risk Log is split in three parts according the study elements:

- overall risk (preliminary) applicable for all study elements (Table 37-4)
- preliminary risks especially applicable for the 'Probe' (Table 37-5)
- preliminary risks especially applicable for the 'Orbiter' (Table 37-6)
- preliminary risks especially applicable for the 'Lander'(Table 37-7)

The study options related to the mission target (Neptune or Uranus) is discussed separately within this 3 'study element'-tables if needed. If there is no specific planet mentioned then the risk identification is applicable for both mission targets.

Risk no. - Title	**Risk Classi-fication** ------------ Risk ranking	**Risk Context/ Scenario** Hazard Source [**HS**], Hazard [**H**], Hazardous Condition [**HC**]	***.. could endanger ..*** Hazard Target [**HT**] ***.. resulting finally in ..*** Unwanted consequences[**UC**]	**Risk Reduction** **possible Mitigation Measure** (if applicable)	**further remarks**
<u>S</u>tudy					
SI - **Assumptions**	program-matic ------------	**HS** .. study information **H** .. incompleteness of study information **HC1:** Many assumptions* for several options (Probe/ Orbiter/ Lander) had to be made to define a closed study baseline options leading to study uncertainties **HC2:** limited number of sessions for a comprehensive study program (e.g. 2 ½ session dedicated to the orbiter; 1 session dedicated to lander ***.. could endanger ..*** **HT** .. study results in general and completeness of Risk log ***.. resulting finally in ..*** **UC** .. immaturity of the final study baseline an incomplete Risk log *Remark:* * *e.g. lifetime, mass budget, no full mission cruise design, simplified structure, aerodynamic 's, maximum design pressure for probe changed in comparison to former study (PEP) from 100kg to 10bar, ...*		- follow-up studies - arrangement of working groups with NASA	
		related risk: not all risks are identified		**related risk:** **OM1**	
SII - **Open points**	program-matic ------------	**HS** .. study scope **H** .. incompleteness of study scope **HC1:** Impact of micro meteoroids not discussed **HC2:** Impact of the collision with ring particles of Jupiter/ Neptune/ Uranus) not discussed **HC3:** Long-lead items not identified **HC4:** planetary protection for combined mission elements (Orbiter+Lander) not discussed **HC5:** TRL readiness * **HC6:** safety aspects not discussed during study ***.. could endanger ..***		- consideration of open points in follow-up studies	

Risk **no.** **-** **Title**	**Risk Classi-fication** ------------ -- Risk ranking	**Risk Context/ Scenario** Hazard Source [**HS**], Hazard [**H**], Hazardous Condition [**HC**]	 ***.. could endanger ..*** Hazard Target **[HT]** ***.. resulting finally in ..*** Unwanted consequences[**UC**]	**Risk Reduction** **possible Mitigation Measure** (if applicable)	 **further remarks**
		HT .. study results in general and completeness of Risk log ***.. resulting finally in ..*** **UC** .. immaturity of the final study baseline and incomplete Risk log *Remark:* ** Technology developments shall be compatible with the programmatic requirement of TRL 6 by end of 2022 (mission adoption) for launch on 13 February 2031*			
		related risk: **HC1/2** -> **DIXb** **HC3** -> **OM1** **HC4** -> **SII-L** **HC5** -> **DI**-P, **DI**-O		**related risk:**	
<u>O</u>verall <u>M</u>gn./PA/Eng.					
OMI - **Inter-face manag ement**	program -matic (schedul e+costs) ------------ --- *	**HS** .. organisational, design and realisation interfaces between space agencies **H** .. interface management **HC1:** differences in program management, project / engineering processes and applicable standards **HC2:** cultural differences ***.. could endanger ..*** **HT** .. schedule + costs ***.. resulting finally in ..*** **UC** .. delay in schedule and cost increase * *Remark:* ** worst case: project cancellation in pre-project phase or during project*		- use of experiences from former projects shared by agencies - specific interface and access requirements	
		related risk:		**related risk:**	
<u>D</u>esign & mission realisation					
DIa - **Radiati on related develo pments**	depen-dability ------------ --	**HS** .. see **DIXa/b / MIX** **H** .. unexpected development needs in pre-project phase **HC:** see **DIXa/b / MIX** ***.. could endanger ..*** **HT** .. project schedule + costs ***.. resulting finally in ..*** **UC** .. schedule delay and cost increase *Remark:* ** applicable especially for the Uranus option because of the closer Jupiter Flyby*		- early identification of development needs in terms of radiation resistance -	
		related risk: **DIXa**		**related risk:**	

<table>
<tr><th rowspan="2">Risk
no.
-
Title</th><th rowspan="2">Risk Classi-fication

Risk ranking</th><th colspan="2">Risk Context/ Scenario</th><th colspan="2">Risk Reduction</th></tr>
<tr><th>Hazard Source [HS],
Hazard [H],
Hazardous Condition [HC]</th><th>.. could endanger ..
Hazard Target [HT]
.. resulting finally in ..
Unwanted consequences[UC]</th><th>possible Mitigation Measure
(if applicable)</th><th>further remarks</th></tr>
<tr><td rowspan="2">DVIIb
-
Ground communication</td><td rowspan="2">protection

(red)</td><td colspan="2">HS .. deep space trajectory

H .. travel time of signals/ information

HC1: Ground communication 2*3.2h data travel time/ 30 AU from the Sun
HC2: Mission operation in critical trajectory phases ** (e.g. Jupiter flyby)

.. could endanger ..
HT .. all mission elements

.. resulting finally in ..
UC .. loss of all mission elements

Remark:
* relatively close Jupiter independent flyby (gravitational assistance); deviations might lead to unrecoverable trajectory deviations

**
-> Neptune/ Uranus orbiter: Jupiter flyby (esp. N.) & orbit insertion & (high S/C autonomy)
-> Neptune/ Uranus probe: entry (high S/C autonomy)</td><td>- high S/C autonomy
- pre-definition of contingency procedures in case of trajectory deviations</td><td></td></tr>
<tr><td colspan="2">related risk:
DVIII, DIXb, DIXa-u, MIX</td><td>related risk:</td><td></td></tr>
<tr><td rowspan="2">DVIII
-
Long Life-time</td><td rowspan="2">protection

(red)</td><td colspan="2">HS .. deep space mission targets

H .. degradation/ derating processes

HC1: long cruise to mission targets
HC2: harsh space environment in terms of (temperature, radiation, ...)

.. could endanger ..
HT .. all mission elements

.. resulting finally in ..
UC .. loss of mission elements</td><td>- design margin
- adequate material and component selection and qualification</td><td></td></tr>
<tr><td colspan="2">related risk:
DVIIb, MIX</td><td>related risk:</td><td></td></tr>
<tr><td>DIXa/b
-
Radiation related loss of mission</td><td>protection/ perfor-mance
----
(red) / (yellow)</td><td colspan="2">HS .. radiation background* on cruise (especially Jupiter flyby**)

H .. interaction with on-board components

HC1: various energy levels/ particles of radiation background leading to short-term effects (SEE) and long-term effect (derating)
HC2: shielding is contributing to mass budget

.. could endanger ..
HT .. all mission elements/ science return(instruments)

.. resulting finally in ..
UC .. loss of mission/ science return</td><td>- radiation resistant component design
of components and equipment
- adequate FDIR concept with prepared contingency measurements</td><td></td></tr>
</table>

Risk no. - Title	Risk Classi-fication ------------ Risk ranking	Risk Context/ Scenario: Hazard Source [HS], Hazard [H], Hazardous Condition [HC]	.. could endanger .. Hazard Target [HT] .. resulting finally in .. Unwanted consequences[UC]	Risk Reduction: possible Mitigation Measure (if applicable)	further remarks
		Remark: ** Shielding against high energetic galactic radiation is almost not possible -> SEE* *** TIDL ratio (krad) between Nep-/ Ura-option for the Jupiter flyby is approximately 1/8 (TBC)*			
		related risk: **DIXb, DIXa-u, MIX**		**related risk:**	
DXX - **Launcher**	program-matic ------------	**HS** .. launcher **H** .. availability of launcher and information about launch load **HC1:** SLS is under development * **HC2:** 1st unmanned flight mid ***.. could endanger ..*** **HT** .. cost + schedule ** ***.. resulting finally in ..*** **UC** .. increase of cost + schedule delay *Remark:* ** 1st unmanned flight mid 2020* *** fixed launch window (2029..34))*		- use of commercial launcher e.g. Falcon9 Heavy -	
		related risk:		**related risk:**	
Launch (including preparation) & LEOP & IOT					
LIIIb - **Release mechanism**	protection ------------	**HS** .. release mechanism * **H** .. interlocking between mission elements **HC:** space environment + long cruise (app. 10years) ***.. could endanger ..*** **HT** .. mission elements/ science return ***.. resulting finally in ..*** **UC** .. loss of mission/ critical reduction of science return *Remark:* ** pyrotechnic and cutters (e.g. for parachute)*		- Tracking in Critical Items List	
		related risk:		**related risk:**	
Cruise and Mission deployment					
MIX - **TID uncertainties**	design ------------	**HS** .. radiation background (especially around IceGiants uncertainties in radiation background) **H** .. knowledge base of radiation background **HC:** the knowledge about the Magnetosphere related radiation (trapped electrons) radiation background based solely on the information collected by the voyager		- R&D for European radiation/plasma/dust model @Neptune - R&D for European radiation/plasma/dust model @Uranus - Embark miniaturized, low power radiation monitor	

Risk no. - Title	Risk Classi-fication ------------ -- Risk ranking	Risk Context/ Scenario: Hazard Source [HS], Hazard [H], Hazardous Condition [HC] .. could endanger .. Hazard Target [HT] .. resulting finally in .. Unwanted consequences[UC]	Risk Reduction: possible Mitigation Measure (if applicable)	further remarks
		missions *.. could endanger ..* **HT** .. element design/ schedule + cost *.. resulting finally in ..* **UC** .. increase of cost + schedule delay *Remark:* ** Potential systematic uncertainties cannot be excluded;* *# Work on-going in new ESA-led model development (SAPPHIRE, TEC-EPS Piers Jiggens et al.) but ESP can be used for now, and is current ECSS prescription* *# Uncertainties in Jupiter trapped energetic electrons flux levels* *- Limited dataset for model building* *- Significant variations observed in dataset* *- Longer term predictions less affected by short term variations*		
		related risk: **DIXa/b** ** *Remark:*** *long terms effect of high radiation might lead to unexpected derating and number of anomalies*	**related risk:** **OC/ OS**	

Other risks

O/P-NASA
Identified by NASA (Ice Giants NASA study 2017, chap.:4.5.4.2)a:
Risks and Concerns for orbiter & probe e.g.
- Orbit Insertion ΔV is sensitive to the orbiter periapsis altitude
- Higher orbiter periapsis provides better relay line-of-sight and longer persistence (lower angular rate relative to probe), but higher NOI ΔV. Shallow FPA reduces probe g-load, but presents challenging telecomm geometry and more TPS mass on the probe due to higher accumulated heat loads.
- Relay antenna must point zenith since the probe rotational phase during EDL cannot be easily predicted. One potential solution that was not explored is to baseline an omnidirectional antenna, or have multiple antennae on the probe.

- Another factor to consider is the time between Probe Entry and NOI. Currently, there are two hours allocated between probe entry and NOI, a critical event. It may be operationally challenging to sequence both the probe relay and NOI on the orbiter within this time window. Increasing the separation will make the geometry more challenging for telecomm.

- Probe-orbiter geometry also needs to deal with issues like uncertainties regarding the Neptune atmosphere and potential signal attenuation. A potential solution would be to perform the NOI burn post periapsis at the cost of increased orbit insertion ΔV.

Risk no. - Title	Risk Classi-fication ------------ -- Risk ranking	Risk Context/ Scenario		Risk Reduction	
		Hazard Source [HS], Hazard [H],	*.. could endanger ..* Hazard Target [HT] *.. resulting finally in ..*	possible Mitigation Measure (if applicable)	further remarks
		Hazardous Condition [HC]	Unwanted consequences[UC]		

P-NASA2
NASA report TEAMX-JPL/ Executive Summary/ Risk - OPTION 3:
Uranus Orbiter concept w/Probe
- 50 kg payload allocation/ - 1 atmospheric probe (previously designed)

- Only 2 hours between Probe entry and UOI, a critical event
  * May be operationally challenging to sequence both the Probe relay and UOI for the Orbiter within this time window.
  * Longer than 2 hours makes the geometry more challenging for telecom.

- May be issues for the relay link margin due to Probe-Orbiter geometry and uncertainties regarding Uranus atmosphere/ potential signal attenuation.

- High g load on the Probe carries some risk.
- Last Probe targeting occurs more than 60 days prior to encounter.
  * Probe carries no propulsion, so it cannot correct trajectory dispersions.
  * Need dispersions small enough to ensure safe entry conditions at Uranus.

- Orbit knowledge requirements for science reconstruction need to be determined.
  * Will drive how the Probe is tracked pre-entry and what telemetry (e.g. IMU) needs to be transmitted with the science data to the Orbiter.
  * The latter will impact the data budget.

- Mission duration will push systems to their operating lifetimes.
- Science planning risk
  * Relative velocities between Orbiter and Uranus' satellites will be high.
    # Flybys occur near periapse

- Collision avoidance with Uranus' rings needs to be considered.
- Uranus stays close to the range of solar conjunction (~4-5 deg)
  * Doppler measurements may have increased noise levels.

- Running the Orbiter power bus to the SEP stage makes for a more complex electronics design and adds cabling.
  * Higher risk than adding a battery on the SEP stage.
  * Chose this to minimize SEP stage mass.

- eMMRTG still needs some development.
  * May cause a schedule slip.
  * Performance may degrade at a higher rate than currently predicted.
  * RTG production rate (~8 years required for 3 eMMRTG) may be too low with respect to requirements

- ROSA solar array qualification carries some risk.
- RTG waste heat recovery design robustness
  * Approach is highly configuration-dependent and may have high hidden development costs.
  * Less expensive on paper, but the actual implementation could be more expensive than an active system.
- Component development for both propulsion subsystems
  * NEXT development for SEP
  *Large bi-prop engines for chemical

Table 37-4: Risk Log applicable for more than one or all study elements

<table>
<tr><th>Risk no. - Title</th><th>Risk Classi-fication ------------ Risk ranking</th><th>Risk Context/ Scenario
Hazard Source [HS], Hazard [H], Hazardous Condition [HC]</th><th>.. could endanger .. Hazard Target [HT]
.. resulting finally in ..
Unwanted consequences[UC]</th><th>Risk Reduction
possible Mitigation Measure (if applicable)</th><th>further remarks</th></tr>
<tr><td colspan="6">Study</td></tr>
<tr><td>SII-P
-
Open points</td><td>program-matic
------------</td><td colspan="2">HS .. study scope
H .. incompleteness of study scope
HC: open study points *
.. could endanger ..
HT .. study results in general and completeness of Risk log
.. resulting finally in ..
UC .. immaturity of the final study baseline and incomplete Risk log
Remark:
*
- Consolidation of Heat Fluxes and Loads
- TPS shall be sized to withstand the aerothermodynamics loads and a complete design iteration shall be performed
- TPS Material Trade-Offs shall be performed (see also OMII.2-P)
- Dynamic pressure impact on structural mass shall be:
duly assessed
- Probe stability shall be ensured
- Payload mass could be revisited
- a FPA = -18 deg could be considered
related risk:
HC1/2 -> DIXb</td><td colspan="2">- consideration of open points in follow-up studies **
Remark:
** A dedicated Probe study is highly recommended (the probe is not just a delta with respect PEP):
- Operational timeline of the payload and scientific requirements shall be clearly identified as both pressure level and scientific measurements duration have an impact on the design
- Black-out during entry and descent (0 deg elevation for 10 minutes) shall be addressed
- 50 g effects on the instruments shall be checked
- Radiation effects on the payload shall be assessed
- DHS mass ok 1 kg shall be revisited as it might be optimistic
- DM Structural mass shall be carefully computed
- Pressure Load on FS shall be looked at
Hot Structure need for the FS (no significant benefit for the FS at -35 deg FPA)
related risk:</td></tr>
<tr><td colspan="6">Overall Mgn./PA/Eng.</td></tr>
<tr><td>OMII.1-P
-
RHU integration</td><td>program-matic
------------</td><td colspan="2">HS .. RHU
H .. integration in probe
HC: integration at non-European location
.. could endanger ..
HT .. schedule + cost
.. resulting finally in ..
UC .. schedule delay/ increase of cost
Remark:
* 31 RHUs are considered necessary to survive 20 day coast phase
related risk:
OMI</td><td>- consideration of adequate accessibility requirements which allow European stuff to follow the probe integration in US
- consideration of additional time slot
related risk:</td><td></td></tr>
</table>

Risk no. - Title	Risk Classi-fication ------------ -- Risk ranking	Risk Context/ Scenario: Hazard Source [HS], Hazard [H], Hazardous Condition [HC]	.. could endanger .. Hazard Target [HT] .. resulting finally in .. Unwanted consequences[UC]	Risk Reduction: possible Mitigation Measure (if applicable)	further remarks
OMII.2-P - **TPS testing**	program-matic ------------ --	**HS** .. TPS shields **H** .. material availability in Europe **HC1:** new development is needed for TPS * **HC2:** no test facility is existing worldwide; testing will only be possible at sample level ** ***.. could endanger ..*** **HT** .. schedule + costs ***.. resulting finally in ..*** **UC** .. increase of cost + delay in schedule *Remark:* * *mass increases against PEP study by 50%* ** *However the risks will be at least partly absorbed by the big margins in design*		- consideration in ESA development programs	
		related risk: **DI-P**		**related risk:**	

Design & mission realisation

Basis for the probe design was the former CDF study PEP with the following general limitations:

1.) the results of the PEP CDF study had to be screened/ updated because of the requested time of science operation of 1.5h (PEP 1h); this was leading to redesign of the probe and parachute dimension:
- probe diameter increased by 10cm to 138cm
 see SI-p (design uncertainties) and mass increase with the possibility to contribute to DII-p (mass/ volume budget);
- parachute design changed complexly from one-stage-parachute system to a two-stage-parachute system and a changed sequence of the release of the back and front heat shield (see DIa-p)

2.) In comparison to the PEP study the operational design pressure could be reduced to 10bar (PEP 100bar) with two effects:
- the probe structure/ pressure vessel could be designed with for a lower pressure -> mass reduction; however a redesign in a very short study period (study uncertainty .. see SI1+2)
- the operation in higher atmosphere (up to 10 bar) leads to lower heat flux** during entry at a given FPA, however the TPS mass increase with respect to PEP related to a current better knowledge of the materials produced an increase in mass, which – in turn – translated into heat flux increase

3.) heat flux might get the leading design risk (**OMII.2-p**)
- the higher the FPA the bigger the heat flux and the lower heat load
- the lower the flight path angle (low change of attitude) the lower the heat flux -35deg.->64 | -18deg.->33MW/m2
 , but the higher the heat loads
 -> TPS shield mass at -35deg. FPA -> Ablator 25mm 81kg | (low HL is good!) -18deg. -> 35mm +32kg |
 - the material of the TPS shield has to be optimised the balance based on head load & heat flux
 - a higher mass will increase load on parachute -> bigger parachute

4.) the Probe is mass ~345kg (study baseline) still compliance with the informative mass budget (350kg) but might become critical with further specification of the mission/ Probe design; PEP design 315kg

Risk no. - Title	Risk Classi-fication ------------ Risk ranking	Risk Context/ Scenario Hazard Source [**HS**], Hazard [**H**], Hazardous Condition [**HC**] *.. could endanger ..* Hazard Target **[HT]** *.. resulting finally in ..* Unwanted consequences[**UC**]	Risk Reduction possible Mitigation Measure (if applicable)	further remarks
DI-P - **TRL status**	program-matic ------------	**HS** .. S/C components/ subsystems **H** .. TRL (Technological Readiness) **HC:** TRL > 6 for all components* at the time of mission adoption end Phase A/B1 ***.. could endanger ..*** **HT** .. schedule + costs ***.. resulting finally in ..*** **UC** .. increase of cost + delay in schedule *Remark:* * *No TRL info's for instruments/ Data Handling (DCM/ CMOS)/ PWR(PCDU)/ structure* *Low TRL COM (UHF_TX/TPS(Back &Front shield)**/parachutes**** ** *actually no test facility is existing worldwide; testing will only be possible at sample level; However the risks will be at least partly absorbed by the big margins in design;* *Furthermore a mass increases against PEP study by 50% has to be considered, related to a more mature knowledge on the material* *** *overall discussion notes about parachute design/deployment* *- changed pressure/ attitude to open parachute (to PEP) will lead to change of parachute (2 needed) -> mass/ volume increase) incl. cost impact* *- CSG -> open parachute as early as possible -> supersonic parachute?* * *Is there the available volume for a much bigger parachute system?* * *A bigger parachute suffers (and generate) much bigger loads requiring stronger and (much) heavier material* * *Need to check when the parachute needs to be deployed to be ready to acquire data at 1 bar* *- study baseline ended up with a 2 times bigger 2stage parachute (surface/ mass)in comparison PEP (1 stage parachute)* *Further design uncertainties:* * *What after reaching 10 bar? Keep descending under parachute? Free fall?* * *Thermal boost before entry to warm up pyros?*	- early consideration in ESA development programs - procurement on non-European market	
		related risk:	**related risk:**	
DII-P - **Para-chute supplier**	program-matic ------------	**HS** .. procurement of parachute **H** .. availability of former parachute supplier * **HC:** the European parachute supplier is economical not stable ***.. could endanger ..*** **HT** .. schedule + costs ***.. resulting finally in ..*** **UC** .. increase of cost + delay in schedule *Remark:* * *PAS procurement (EXM had problems with supplier bankruptcy)*	- procurement as early as possible - qualification of new European supplier - procurement on non-European market	
		related risk: **LIIIb-P, DI-P**	**related risk:**	

Risk no. - Title	Risk Classi-fication ------------ -- Risk ranking	Risk Context/ Scenario: Hazard Source [HS], Hazard [H], Hazardous Condition [HC]	.. could endanger .. Hazard Target [HT] .. resulting finally in .. Unwanted consequences[UC]	Risk Reduction: possible Mitigation Measure (if applicable)	further remarks
Launch (including preparation) & LEOP & IOT					
LIIIb-P - Para-chute deploy-ment	protection ------------ --	**HS** .. parachute system **H** .. deployment of parachute system **HC1:** space environment + long cruise **HC2:** adoption of military used design which is not space used so far ***.. could endanger ..*** **HT** .. probe/ science return ***.. resulting finally in ..*** **UC** .. loss of probe/ science return		- -	
		related risk: **DII-P**		**related risk:**	
Cruise and Mission deployment					
CIb-P - Instrument deployment/ Data collection/ transmission	per-formance ------------ --	**HS** .. entry conditions **H** .. limited time frame **HC1:** short instrument deployment period after separation from orbiter 2h **HC2:** very short deployment period after release of back/ front shields (5min) **HC2:** fixed science period during entry (90min) incl. data return to orbiter ***.. could endanger ..*** **HT** .. science return ***.. resulting finally in ..*** **UC** .. loss of science return *Remark:* ** Data is generated in 90 mins at 2 kbps (significantly larger data volume than Galileo of ~factor 3)*		- advanced FDIR concept for fast reaction in case of anomalies incl. comprehensive plausibility checks of incoming GNC system - full redundancy for data handling system	
		related risk: **LIIIb-P, DI-P**		**related risk:**	
Other risks					

P-NASA1
Identified by NASA (Ice Giants NASA study 2017 4.7.4.3/ A5.5):
Due to the fact that -18° shallow trajectory has certain drawbacks, such as higher heat load leading to higher TPS mass, a potential compromise in communications (link visibility), as well as a greater risk of skip out, however it would enable the usage of European test facilities, in the future it will be worthwhile to investigate trajectories at -18° FPA and attain a more optimum design …

If the specifications and performance parameters of the instruments and thermal protection system (TPS) materials used in this study are valid when the mission is implemented, the prograde probe entry adopted for this study appears feasible. But the modified entry trajectory adopted here requires a steeper-than-usual entry flight path angle (FPA). This produces higher peak heating rates and significantly higher inertial loads than a shallower FPA. It is possible that as this mission concept

Risk no. - Title	Risk Classi-fication ------------ -- Risk ranking	Risk Context/ Scenario		Risk Reduction	
		Hazard Source [**HS**], Hazard [**H**], Hazardous Condition [**HC**]	***.. could endanger ..*** Hazard Target **[HT]** ***.. resulting finally in ..*** Unwanted consequences[**UC**]	**possible Mitigation Measure** (if applicable)	**further remarks**

evolves, changes in approach circumstances, less-than-anticipated performance by various components, or other unforeseen drivers could force the prograde approach to an entry that is riskier than the project is willing to accept.

<u>NASA reportTEAMX-JPL/ Executive Summary/ Risk - OPTION 4:</u>
Option 2: Uranus Orbiter Variant Concept
- 150 kg payload allocation/ - No atmospheric probe/ - No crosslink telecom hardware

- Mission duration will push systems to their operating lifetimes.
- Science planning risk
  * Relative velocities between Orbiter and Uranus' satellites will be high.
    # Flybys occur near periapse

- Collision avoidance with Uranus' rings needs to be considered.

- Uranus stays close to the range of solar conjunction (~4-5 deg)
  * Doppler measurements may have increased noise levels.

- Running the Orbiter power bus to the SEP stage makes for a more complex electronics design and adds cabling.
  * Higher risk than adding a battery on the SEP stage.
  * Chose this to minimize SEP stage mass.

- eMMRTG still needs some development.
  * May cause a schedule slip.
  * Performance may degrade at a higher rate than currently predicted.

- ROSA solar array qualification carries some risk.
- RTG waste heat recovery design robustness
  * Approach is highly configuration-dependent and may have high hidden development costs.
  * Less expensive on paper, but the actual implementation could be more expensive than an active system.

- Component development for both propulsion subsystems
  * NEXT development for SEP
  * Large bi-prop engines for chemical

- Sun sensor performance may degrade past Saturn.
  * May impact safe mode used during star tracker outage.

Table 37-5: Risk Log specifically applicable for the 'Probe'

Risk no. - Title	Risk Classi-fication ----------- - Risk ranking	Risk Context/ Scenario: Hazard Source [HS], Hazard [H], Hazardous Condition [HC]	.. could endanger .. Hazard Target [HT] .. resulting finally in .. Unwanted consequences[UC]	Risk Reduction: possible Mitigation Measure (if applicable)	further remarks
Study					
SII-O - **Open points**	program-matic ------------	**HS** .. study scope **H** .. incompleteness of study scope **HC:** Open points * ***.. could endanger ..*** **HT** .. study results in general and completeness of Risk log ***.. resulting finally in ..*** **UC** .. immaturity of the final study baseline and incomplete Risk log *Remark:* *** *- RTGs characteristics shall be better understood: Power degradation characteristics (esp. EOL power output)* *Number of RTGs that could actually be made available to an ESA mission element* *- HGA diameter and pointing capability shall be revisited* *- Structural mass is on the low side (~7% of the dry mass) and shall be revisited*		- consideration of open points in follow-up studies ** *Remark:* *** *- Scientific Requirement of downloading 32 Gb data volume at the end of the mission is currently (50 days orbit) looking challenging for the Neptune case* *- For the Uranus case the data volume downloadable with the Neptune design is 53 Gb* *- Payload timeline shall be consolidated, tailoring it such to meet boundary conditions e.g. doing less periapsis science in the shorter orbits (which occur near the end of the mission, when the planetary periapsis should already be well characterized)* *- Dual orbiter accommodation inside the LV trade-off* *- Minimal distance between 2 engines in Neptune (as per current configuration) shall be checked for thermal issues (also attitude control shall be investigated), as they are operated simultaneously for ~ 30 mins* *- Alternative Jupiter flybys for targeting Uranus in different launch dates should be carefully investigated, sensitivity to be understood, and a detailed assessment of the radiation environment should be provided* *- Alternative radiation shielding materials could be traded off to target mass savings* *- Energy excess available during cruise could be considered for science usage* *- Dedicated design for a specific planet could bring optimisation to some of the subsystems*	
		related risk: **CIb,**		**related risk:**	
Overall Mgn./PA/Eng.					
OMII.1-O - **RHU integration**	program-matic ------------	**HS** .. RTG **H** .. integration in orbiter * **HC:** integration at non-European location ***.. could endanger ..*** **HT** .. schedule + cost ***.. resulting finally in ..*** **UC** .. schedule delay/ increase of cost *Remark:* *** *- launch and launch site safety requirements* *- Ground facilities for integration* *- integration through fairing doors*		- consideration of adequate accessibility requirements which allow European stuff to follow the probe integration in US - consideration of additional time slot	
		related risk: **OMI**		**related risk:**	

Risk no. - Title	Risk Classification - Risk ranking	Risk Context/ Scenario: Hazard Source [HS], Hazard [H], Hazardous Condition [HC] / .. could endanger .. Hazard Target [HT] .. resulting finally in .. Unwanted consequences[UC]	Risk Reduction: possible Mitigation Measure (if applicable)	Risk Reduction: further remarks
Design & mission realisation				
DI-O - **TRL status**	program-matic	**HS** .. S/C components/ subsystems **H** .. TRL (Technological Readinesso **HC:** TRL > 6 for all components* at the time of mission adoption end Phase A/B1 ***.. could endanger ..*** **HT** .. schedule + costs ***.. resulting finally in ..*** **UC** .. increase of cost + delay in schedule *Remark:* ** No TRL info's for instruments **/ AOCS(STR)/ PWR(PCDU)/ structure* *Low TRL COM(HGA+ Ka-equipment)* *** deployable boom for magnetometer (Neptune) .. European instrument with high TRL - very heavy*	- early consideration in ESA development programs - procurement on non-European market	
		related risk:	**related risk:**	
DII-O - **RTU Procur-ment avail-ability**	program-matic	**HS** .. RTG **H** .. RTG production capacities **HC1:** relatively high amount of Uranium needed * **HC2:** limited RTG production capacities ***.. could endanger ..*** **HT** .. project ***.. resulting finally in ..*** **UC** .. project cancellation/ delay *Remark:* ** 3 RTUs a 4kg Uranium -> 8 years of actual production rate; 3 RTGs are essential to enable any type of science* ** a dual orbiter scenario this would imply high number of RTGs (6), with following issues to be considered:*	- early start of procurement	
		related risk:	**related risk:**	
DIII-Ou - **Mass budget**	program-matic	**HS** .. mass contribution of subsystems/ components/ media **H** .. mass limit **HC:** high radiation requires adequate shielding additionally to the shielding given by 'usual' structures ***.. could endanger ..*** **HT** .. schedule + costs ***.. resulting finally in ..*** **UC** .. increase of cost + delay in schedule	- optimisation of Jupiter flyby in terms of radiation reduction ((understanding dependency with launch date and increase distance from the planet) - mass reduction in other subsystem domains	

Risk no. - Title	Risk Classi-fication ------------- Risk ranking	Risk Context/ Scenario: Hazard Source [HS], Hazard [H], Hazardous Condition [HC]	.. could endanger .. Hazard Target [HT] .. resulting finally in .. Unwanted consequences[UC]	Risk Reduction: possible Mitigation Measure (if applicable)	further remarks
		related risk: **DXX, DIXa/b**		**related risk:**	
DIV-O - **Power, Thermal**	perfor-mance ------------- (yellow)	**HS** .. RTU **H** .. performance of RTU **HC1:** long cruise period **HC2:** uncertainties about end-of-life performance (+ 15%) ***.. could endanger ..*** **HT** .. science return ***.. resulting finally in ..*** **UC** .. critical reduction of science return		- -	
		related risk:		**related risk:**	
Cruise and Mission deployment					
CIa-O - **cruise accuracy**	protection ------------- (red)	**HS** .. deep space trajectory **H** .. deviations from trajectory **HC1:** several critical manoeuvre has to be performed to reach mission target (Jupiter flyby *, Orbit deflection/ insertion) **HC2:** high manoeuvre accuracy needed to reach mission target **HC3:** limited time windows to perform manoeuvres ***.. could endanger ..*** **HT** .. mission element/science return ***.. resulting finally in ..*** **UC** .. loss of mission element/science return *Remark:* ** relatively close Jupiter independent flyby (gravitational assistance); deviations might lead to unrecoverable trajectory deviations*		- high on-board autonomy - advanced FDIR concept - intermediate health checks	
		related risk:		**related risk:**	
Other risks					

Identified during the Internal Final Presentation:
Minimal distance between 2 engines in Neptune (as per current configuration) shall be checked for thermal issues (also attitude control shall be investigated), as they are operated simultaneously for ~ 30 mins [single thruster to be considered? It is recommended to inject the ExoMars lessons learned]

Identified by NASA (Ice Giants NASA study 2017, chap.:4.5.4.2)a:
Risks and Concerns for orbiter e.g.
- Orbit Insertion ΔV is sensitive to the orbiter periapsis altitude
- Higher orbiter periapsis provides better relay line-of-sight and longer persistence (lower angular rate relative to probe), but higher NOI ΔV. Shallow EFPA reduces probe g-load, but presents challenging telecomm geometry and more TPS mass on the probe due to higher accumulated heat loads.

Risk no. - Title	Risk Classi-fication ----------- - Risk ranking	Risk Context/ Scenario: Hazard Source [HS], Hazard [H], Hazardous Condition [HC]	.. could endanger .. Hazard Target [HT] .. resulting finally in .. Unwanted consequences[UC]	Risk Reduction: possible Mitigation Measure (if applicable)	further remarks
- Relay antenna must point zenith since the probe rotational phase during EDL cannot be easily predicted. One potential solution that was not explored is to baseline an omnidirectional antenna, or have multiple antennae on the probe. - Another factor to consider is the time between Probe Entry and NOI. Currently, there are two hours allocated between probe entry and NOI, a critical event. It may be operationally challenging to sequence both the probe relay and NOI on the orbiter within this time window. Increasing the separation will make the geometry more challenging for telecomm. - Probe-orbiter geometry also needs to deal with issues like uncertainties regarding the Neptune atmosphere and potential signal attenuation. A potential solution would be to perform the NOI burn post periapsis at the cost of increased orbit insertion ΔV.					

Table 37-6: Risk Log specifically applicable for the 'Orbiter'

Risk no. - Title	Risk Classi-fication ----------- - Risk ranking	Risk Context/ Scenario: Hazard Source [HS], Hazard [H], Hazardous Condition [HC]	.. could endanger .. Hazard Target [HT] .. resulting finally in .. Unwanted consequences[UC]	Risk Reduction: possible Mitigation Measure (if applicable)	further remarks
Study					
SII-L - **Open points**	program-matic ----------- --	**HS** .. study scope **H** .. incompleteness of study scope **HC1:** planetary protection not discussed* **HC2:** other open points and findings. ** ***.. could endanger ..*** **HT** .. study results in general and completeness of Risk log ***.. resulting finally in ..*** **UC** .. immaturity of the final study baseline and incomplete Risk log *Remark:* ** for the Neptune moon Triton the same conditions in terms of the origin of life like for the moons Enceladus or Europe applies; therefore to search there for signs of life the lander should be decontaminated from terrestrial forms of life* *** In order to land ~ 10 kg payload during Triton fly-by, Lander wet mass is 2.0-2.5 tons*		- consideration of open points in follow-up studies *** - further technical developments **** *Remarks:* ***** *Dedicated study shall address the major issues of:* *- Plume impingement contamination during descent* *- Landing gear sizing coping with unknown surface characteristics* *- Lander release strategy and impact on lander and carrier* *- Visibility (rage, elevation) of the carrier for communication relay* ****** *Propulsion .. throttled / pulsed propulsion capabilities in a closed-loop GNC system for the final descent manoeuvre* *GNC .. Dual-use LIDAR, hazard-detection-avoidance and altimeter* *Power .. Combined avionics/power unit approach, as a "Minavio" concept* *Operation .. Array of up to 4 ground stations, with 2 receiver-only antennas, and 2 Transmitting + receiving*	

Risk no. - Title	Risk Classi-fication ------------- Risk ranking	Risk Context/ Scenario: Hazard Source [HS], Hazard [H], Hazardous Condition [HC]	.. could endanger .. Hazard Target [HT] .. resulting finally in .. Unwanted consequences[UC]	Risk Reduction: possible Mitigation Measure (if applicable)	further remarks
		related risk: **HC1/2** -> **SII** (HC4)		**related risk:**	
Cruise and Mission deployment					
CIc.1-L - Landing site	per-fomance -------------	**HS** .. landing site **H** .. selection of landing site **HC1:** landing site should be close to ice plume areas **HC2:** limited possibilities to monitor moon surface from orbiter	***.. could endanger ..*** **HT** .. science return ***.. resulting finally in ..*** **UC** .. major reduction in science return (landing site away from ice plume fallout areas)	- release of lander in later phase after several moon flybys of the orbiter	
		related risk:		**related risk:** **DIXb, MIX**	
CIc.2-L - Instrument deployment/ Data collection/ trans-mission	protection -------------	**HS** .. lander descent **H** .. limited time frame **HC1:** the descent manoeuvre has to be performed fully automatically incl. handling of non-rational data (plausibility checks) and anomalies) **HC2:** descent sequence is extreme short without very limited correction possibilities also for the on-board FDIR	***.. could endanger ..*** **HT** .. lander/ science return ***.. resulting finally in ..*** **UC** .. loss of lander/ science return	- advanced FDIR concept for fast reaction in case of anomalies incl. comprehensive plausibility checks of incoming AOCS system - full redundancy for data handling system	
		related risk:		**related risk:**	
CId-L - landing conditions	perfor-mance -------------	**HS** .. landing surface **H** .. landing condition of site **HC1:** stability of landing ground is highly unknown* **HC2:** admixtures in the ice (plumes might come from underground oceans with active geology)	***.. could endanger ..*** **HT** .. science return ***.. resulting finally in ..*** **UC** .. loss of science return *Remark:* ** the preferred landing site is next to the ice plumes to search for signs of life in the assumed underground ocean; however, under low gravity and extreme low*	- -	

Risk no. - Title	Risk Classi-fication ---------- - Risk ranking	Risk Context/ Scenario		Risk Reduction	
		Hazard Source [HS], Hazard [H], Hazardous Condition [HC]	*.. could endanger ..* Hazard Target [HT] *.. resulting finally in ..* Unwanted consequences[UC]	possible Mitigation Measure (if applicable)	further remarks
		temperature it is possible that the ice surface/ structure might not carry the lander; in case the lander would drown in the ice the RF contact might be lost due to minerals in the ice coming as well from the underground ocean			
		related risk:		**related risk:**	

Table 37-7: Risk Log specifically applicable for the 'Lander'

37.5.1 Risk Log General Conclusions

- Very high risks and high risks are typical of a phase A project. Areas with lack of definition or little previous experience pose a priori more risk to the mission and therefore are the ones with more risk reduction potential
- Experience shows that all risk items with a critical risk index (red, orange area) must be analysed and proposals for risk treatment actions elaborated
- In the end, ideally all risk items should achieve a level of justifiable acceptance
- The risk management process should be further developed during the project definition phase in order to refine the risk identification/analysis and provide evidence that all the risks have been effectively controlled.

37.6 Risk Log Specific Conclusions and Recommendations

The study has to be seen as a preliminary feasibility study for several mission element-options (Orbiter or Probe or Lander) dedicated to the Ice Giants Neptune and Uranus in the frame of a NASA opportunity mission.

Therefore a 'Risk identification' was seen as more suitable than an extended Risk assessment.

The scientific objectives of the Ice Giants mission:

- To contribute to the observation one of the Ice Giants and its moons (Neptune+Triton/ Uranus)**

One of the main focus of this study was the identification of early development needs which are also subject of the Risk Log'- Table 37-4 to Table 37-7.

The Ice Giants study covers a very complex mission with many mission elements (probe or orbiter or lander), mission phases (interplanetary trajectory with a Jupiter flyby, orbit insertion and landing) and options (mission target: Neptune or Uranus) where only limited practical ESA-internal experiences are available related to platform components (e.g. the use/ integration of RTG, RHU and the use of high-radiation resistant Star tracker) and payload instruments resistant against high radiation. Naturally many risks were identified (see Table 37-4 to Table 37-7). However, such comprehensive risk portfolio is in general not unusual for a deep space mission and especially not in case of several mission elements and options.

The uncertainty about the identified risks is relatively high caused by:

- The limited time-frame of the study , the amount of assumptions which had to be made related to the incomplete knowledge of the NASA mission requirements/ assessment (e.g. trajectory conditions for cruise/ orbit insertion, availability of nuclear components) .. **SI**
- The weak data base related to the mission targets Neptune/ Uranus (e.g. Magnetosphere / radiation environment); Furthermore results from a former probe study (PEP) were 'only' partially applicable (e.g. change of entry trajectory) leading to re-designed (e.g. parachute system) .. e.g. **MIX** contributing also to **DIXa/b**,
- Several open points which has still to be analysed in follow up studies (**SII**, **SII**-P/-O/-L) or working groups

Special early development needs were identified in the area of:

- The Probe parachute system .. **DII**-P**, LIIIb**-P
- The Thermal Protection System of the Probe .. **OMII.2**-P

and applicable for all study elements:

- The Data transmission Ka-Band equipment .. **DI**-P/-O
- The instruments .. **DI**-P/-O

Further development needs are addressed in (**SII**, **SII**-P/-O/-L).

Nevertheless the study did not identify 'show stoppers' for ESA contribution. However, the design of the Uranus-orbiter (**DIII**-Ou) might increase in mass limit of due to high radiation during Jupiter flyby. Nevertheless the mass budget is given for orientation only and has still to be verified.

Further delta studies are needed for all study/ mission elements to consolidated the Risk identification and to perform a Risk assessment.

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38 PROGRAMMATICS/AIV

38.1 Requirements and Design Drivers

The main requirements and design drivers for the Ice Giants project from a programmatic point of view are:

SubSystem Requirements		
Req. ID	**Statement**	**Parent ID**
PROG-010	The Program Milestones on ESA and NASA side shall be consistent with a logic where the System Level (NASA) PDR is first performed, with Element level (ESA) PDR following, then all Subcontractors PDR's . CDR logic shall be with Subcontractors CDR's first, followed by Element level (ESA) CDR, concluding then with the System (NASA) CDR.	
PROG -020	Any deviation from the above milestones logic (PROG-010) shall be agreed at the level of Joint NASA/ESA Program Board	
PROG -030	The ESA delivery date for its FM Elements shall be compatible with a mission launch date fixed on February 2031. The ESA Elements delivery due date is currently fixed on mid October 2029, including an ESA schedule contingency of 6 months.	
PROG -040	In case the Heat Shield TPS of a Planetary Probe is procured by ESA, the verification method for the Heat Shield Thermal Protection System, against the planetary entry heat flux, shall be analysis, supported by appropriate sample level testing by the available European test facilities.	
PROG -050	In case the Parachute System of a Planetary Probe is procured by ESA, a Subcontractor Survey shall be launched at PRR in order to select the Parachute System supplier well in advance of mission adoption. This is a condition to the success of the Program.	
PROG -060	Element interfaces with the NASA system, namely data and power, shall be verified by test with a NASA provided System Interface Simulator, reproducing the required interface characteristics for realistic testing.	
PROG -070	Power compatibility of NASA System with ESA Orbiter shall be verified by test.	

38.2 Assumptions and Trade-Offs

Assumptions	
1	It is assumed that the System Composite AIT activities at NASA, after ESA FM Elements delivery, will last 13 months, including 3 months of NASA contingency.
2	It is assumed that the launch campaign, under the responsibility of NASA, will last 3 months.
3	It is assumed that for the Ice Giants ESA Orbiter, an aggressive model philosophy

Assumptions	
	with AVM (Avionics Verification Model) and PFM Element level models can be applied; due to the good heritage European Industry have with this class of spacecraft.
4	An Element level STM (structural and thermal model) is not currently requested, but it could be accommodated within the margins of the presented schedule, if requested by NASA as deliverable model to System AIT, and as technical risk reduction method.
5	An Element level EM may be required in place of the AVM, depending on NASA System test level requirements and program technical risk reduction requirements. This is to be subject of a further investigation with NASA.

38.3 Technology Requirements

The Technology Readiness Levels (TRL) present a systematic measure, supporting the assessments of the maturity of a technology of interest and enabling a consistent comparison in terms of development status between different technologies.

The product tree for Ice Giants, as established in the CDF workbooks, is shown in Table 38-2 (probe) and Table 38-2 (orbiter). It identifies for each subsystem the associated equipment, their quantity and their TRL as far as available.

Category	Owner	Name	n_items	shape	TRL
Elements	**SYE**	**Probe**			
Subsystems	**COM**	**Communications Subsystem**			
Equipment	COM	UHF Radio Frequency Distribution Network			9
Equipment	COM	UHF Low Gain Antenna			9
Equipment	COM	UHF Solid State Power Amplifier			8
Equipment	COM	UHF Solid State Power Amplifier #2			8
Equipment	COM	UHF Transmitter			4
Equipment	COM	UHF Transmitter #2			4
Equipment	COM	UHF Patch LGA			5
Subsystems	**DH**	**Data-Handling Subsystem**			
Equipment	DH	Computer and Data Management Probe #2			TBD
Subsystems	EDL	Entry, Descent and Landing Subsystem			
Equipment	EDL	Main parachute			3
Equipment	EDL	Pilot chute			3
Subsystems	**GNC**	**Guidance, Navigation and Control Subsystem**			
Equipment	GNC	LN200S #1			9
Equipment	GNC	LN200S #2			9
Equipment	GNC	PAS Switch #1			TBD
Equipment	GNC	PAS Switch #2			TBD

Subsystems	INS	Instruments Subsystem			
Equipment	INS	Atmospheric Structure Instrument			TBD
Equipment	INS	Camera-Radiometer			TBD
Equipment	INS	Mass Spectrometer			TBD
Equipment	INS	Photometer			TBD
Equipment	INS	USO-Doppler			TBD
Subsystems	**MEC**	**Mechanisms Subsystem**			
Equipment	MEC	Back Shell Separation Mechanism [probe side]			9
Equipment	MEC	Back Shell Separation Mechanism [DM side]			9
Equipment	MEC	Front Shield Seperation Mechanism [DM side]			9
Equipment	MEC	Front Shield Seperation Mechanism Mec [probe side]			9
Equipment	MEC	Spin Ejection Mechanism [probe side]			9
Equipment	MEC	Mortar Parachute Pyro Cutter			9
Subsystems	**PWR**	**Power Subsystem**			
Equipment	PWR	Batteries	4		6
Equipment	PWR	Power Conditioning & Distribution (PCDU)			5
Subsystems	**STR**	**Structures Subsystem**			
Equipment	STR	Back Shield Cold Structure			TBD
Equipment	STR	Back Shield To DM I/F Brackets	3		TBD
Equipment	STR	Back Shield Stiffening Ribs	3		TBD
Equipment	STR	DM Mounting Platform #1			TBD
Equipment	STR	DM Mid-Section Ring			TBD
Equipment	STR	DM Shell			TBD
Equipment	STR	DM Main Parachute Supporting Structure #1)	3		TBD
Equipment	STR	Front Shield Cold Structure			TBD
Equipment	STR	Front Shield I/F Brackets	3		TBD
Equipment	STR	Front Shield Separation Ring			TBD
Subsystems	**TC**	**Thermal Control Subsystem**			
Equipment	TC	Backcover MLI			6
Equipment	TC	Frontshield MLI			6
Equipment	TC	RHUs	31		TBD
Equipment	TC	RHUs support	31		TBD
Equipment	TC	Pressure vessel insulation			TBD
Subsystems	**TP**	**Thermal Protection Subsystem**			
Equipment	TP	Backcover Ablator			5
Equipment	TP	Frontcover Ablator			4
Equipment	TP	Backcover Hot structure			

Equipment	TP	Frontcover Hot structure			TBD
Equipment	TP	Backcover insulation			TBD
Equipment	TP	Frontshield insulation			TBD
Equipment	TP	Heatshield instruments			TBD

Table 38-1: Ice Giants product tree – probe at IFP

Category	Owner	Name	n_items	shape	TRL
Elements	**SYE**	**Spacecraft**			
Equipment	AOGNC	Attitude, Orbit, Guidance, Navigation Control Subsystem			
Equipment	AOGNC	IMU Airbus Astrix 1090A #1			9
Equipment	AOGNC	IMU Airbus Astrix 1090A #2			9
Equipment	AOGNC	NavCam #1			9
Equipment	AOGNC	NavCam #2			9
Equipment	AOGNC	RW Honeywell HR04 #1			7
Equipment	AOGNC	RW Honeywell HR04 #2			7
Equipment	AOGNC	RW Honeywell HR04 #3			7
Equipment	AOGNC	RW Honeywell HR04 #4			7
Equipment	AOGNC	STR Sodern Hydra JUICE Electronics Unit #1			9
Equipment	AOGNC	STR Sodern Hydra JUICE Electronics Unit #2			9
Equipment	AOGNC	STR Sodern Hydra JUICE Optical Head #1			9
Equipment	AOGNC	STR Sodern Hydra JUICE Optical Head #2			9
Equipment	AOGNC	GYRO Selex Galileo Sireus			7
Subsystems	**COM**	**Communications Subsystem**			
Equipment	COM	High Gain Antenna (HGA)			9
Equipment	COM	Ka-Band Electronic Power Conditioning			2
Equipment	COM	Ka-Band Electronic Power Conditioning – Redundant			2
Equipment	COM	Ka-Band Traveling Wave Tube			2
Equipment	COM	Ka-Band Traveling Wave Tube - Redundant			2
Equipment	COM	Low Gain Antenna – LHCP			9
Equipment	COM	Low Gain Antenna – RHCP			9
Equipment	COM	Radio Frequency Distribution Network			9
Equipment	COM	X-Band Electronic Power Conditioning			9
Equipment	COM	X-Band Electronic Power Conditioning – Redundant			9
Equipment	COM	X/X/Ka-Band Transponder – Redundant			9
Equipment	COM	X/X/Ka-Band Transponder			9
Equipment	COM	X-Band Traveling Wave Tube			9
Equipment	COM	X-Band Traveling Wave Tube - Redundant			9
Subsystems	**CPROP**	**Chemical Propulsion Subsystem**			
Equipment	CPROP	Biprop FillDrain Valves	9		9

Equipment	CPROP	Biprop Filters	4		9
Equipment	CPROP	Biprop LP Transducer	4		9
Equipment	CPROP	Biprop Latch Valves	4		9
Equipment	CPROP	Biprop Non-Return Valves	4		9
Equipment	CPROP	Biprop Pipes			9
Equipment	CPROP	Biprop Thruster Main	2 (/1)		7
Equipment	CPROP	Biprop Pressure Regulator	2		9
Equipment	CPROP	Biprop Pressurant Tank	2		9
Equipment	CPROP	Biprop Propellant Tank	4		9
Equipment	CPROP	Biprop SMA Valves	2		9
Equipment	CPROP	Biprop RCS Thrusters	16		9
Equipment	CPROP	Biprop HP Latch Valve			9
Equipment	CPROP	Biprop HP Transducer			9
Equipment	CPROP	Biprop Pressurant Tank (small) (only for Neptune)			TBD
Equipment	CPROP	Biprop Pyro Valves	4		9
Subsystems	**DH**	**Data-Handling Subsystem**			
Equipment	DH	Remote Interface Unit Centralised			5
Equipment	DH	Remote Interface Unit Decentralised			5
Equipment	DH	Computer and Data Management Unit #1			6
Equipment	DH	Computer and Data Management Unit #2			6
Subsystems	**INS**	**Instruments Subsystem**			
Equipment	INS	Camera			TBD
Equipment	INS	Imaging Spectrometer			TBD
Equipment	INS	Ion and Neutral Mass Spectrometer #1			TBD
Equipment	INS	Ion and Neutral Mass Spectrometer #2			TBD
Equipment	INS	Ion and Neutral Mass Spectrometer #3			TBD
Equipment	INS	Instrument Ka-Band Electronic Power Conditioning			TBD
Equipment	INS	Instrument Ka Band Traveling Wave Tube			TBD
Equipment	INS	Ka-band Trransponder			TBD
Equipment	INS	Magnetometer			TBD
Equipment	INS	Microwave radiometer			TBD
Equipment	INS	Ultra Stable Oscillator (USO)			TBD
Subsystems	**MEC**	**Mechanisms Subsystem**			
Equipment	MEC	Deployable magnetometer boom - *Pyros*	*3*		*9*
Equipment	MEC	SEP stage separation [SC side]			TBD
Subsystems	**PWR**	**Power Subsystem**			
Equipment	PWR	Batteries	4		7
Equipment	PWR	Enhanced Multi-Mission RTG (EMMRTG)	3		4
Equipment	PWR	External Power Shunt	1		5

Equipment	PWR	Power Conditioning & Distribution Unit (PCDU)	1		4
Equipment	PWR	Resisitive Power Shunt	3		5
Subsystems	**RAD**	**Radiation Subsystem**			
Equipment	RAD	Radiation Monitor NGRM			7
Subsystems	**STR**	**Structures Subsystem**			
Equipment	STR	Assembly Panels			TBD
Equipment	STR	Bottom Panel			TBD
Equipment	STR	CPROP Tank Deck			TBD
Equipment	STR	Module Collars			TBD
Equipment	STR	Shear Panels			TBD
Equipment	STR	Top Panel			TBD
Equipment	STR	Tube Rings			TBD
Equipment	STR	Tank Supporting Struts			TBD
Equipment	STR	Tank Supporting Tube			TBD
Subsystems	**TC**	**Thermal Control Subsystem**			
Equipment	TC	Black Paint			TBD
Equipment	TC	Louvres			TBD
Equipment	TC	MLI external (22-layer)			6
Equipment	TC	MLI HGA (10-layer)			6
Equipment	TC	MLI internal (10-layer)			6
Equipment	TC	MLI RTG radiative shield			TBD
Equipment	TC	Radiator SSM-tape			TBD
Equipment	TC	White Paint			TBD
Equipment	TC	Heaters			TBD

Table 38-2: Ice Giants product tree – orbiter at IFP

TRL 4 has been assumed for all the products where information on TRL was missing. Exceptions are: the parachute with TRL=3; instruments with TRL= TBD; Ka equipment with TRL=2. With 4 years before the SRR at the time of writing, it is considered credible to be able developing all equipment having TRL=4.

The TRL definitions from ISO are shown in Table 38-3:

TRL	ISO Definition	Associated Model
1	Basic principles observed and reported	Not applicable
2	Technology concept and/or application formulated	Not applicable
3	Analytical and experimental critical function and/or characteristic proof-of concept	Mathematical models, supported e.g. by sample tests
4	Component and/or breadboard validation in laboratory environment	Breadboard

TRL	ISO Definition	Associated Model
5	Component and/or breadboard critical function verification in a relevant environment	Scaled EM for the critical functions
6	Model demonstrating the critical functions of the element in a relevant environment	Full scale EM, representative for critical functions
7	Model demonstrating the element performance for the operational environment	QM
8	Actual system completed and "flight qualified" through test and demonstration	FM acceptance tested, integrated in the final system
9	Actual system completed and accepted for flight ("flight qualified")	FM, flight proven

Table 38-3: TRL scale

Although a general statement is made, that only technology sufficiently advanced (TRL) to start the Implementation Phase will be proposed, there are TRL as low as 2, 3, 4 and 5 identified.

Table 38-4 shows an indication of the development time depending on the current TRL. According to the European Space Technology Master Plan, to prepare the contractual basis for multi-annual programs it takes about 18 months to reach political agreement on financial ceiling. This has also been included in the table.

TRL	Duration
5-6	4 years + 1.5 year
4-5	6 years + 1.5 year
3-4	8 years + 1.5 year
2-3	10 years + 1.5 year
1-2	12 years + 1.5 year

Table 38-4: TRL – development duration

Assuming, that the development of technology at TRL lower than 6 is already approved and on-going, it can be expected that another 2 years is needed before the implementation phase can start for technologies at TRL 4 and another 4 years for technologies at TRL 3 unless very special effort is made to speed up the development.

38.4 Model Philosophy

The Ice Giants model philosophy is driven by the System Level (NASA), and ESA Elements level will have to agree with NASA the most suitable approach in terms of technical quality, verification effectiveness, schedule constraints and cost affordability. On the basis of these constraints, a model philosophy for the ESA Elements verification by test is proposed.

38.4.1 Orbiter

A model philosophy including an Avionics Verification Model (AVM) and the Protoflight Model (PFM) is proposed. As an option, a Structural and Thermal Model (STM) may be added, but it is currently not considered. See the option paragraphs for additional possibilities, not included today.

38.4.1.1 Orbiter AVM

On the Orbiter AVM the functional verification of the on-board electrical system including SW will be accomplished with a number of Element level tests, which procedures, after successful debug and flawless execution, will be then applied to the PFM Orbiter. The newly delivered versions of the on-board SW will be tested on this model before loading them onto the PFM. The AVM will implement EM quality avionic units (see 38.4.1.2) or elegant breadboard quality, if deemed sufficient. The AVM implementation quality shall allow for conducted EMC test, as pre-qualification under EMC CE-CS environment of the Orbiter Element.

This model allows for System level verification with NASA, with the possibility to interface with a NASA System I/F simulator for data traffic verification and power compatibility verification by test. Note that power quality and compatibility with NASA System will have to be verified by test also on the Orbiter PFM.

38.4.1.2 Option: Orbiter EM

As an option to the Orbiter AVM, and depending on the agreements with NASA, an EM quality Orbiter Model may need to be made available. This model would be completely representative of the flight Orbiter in terms of form, electrical interfaces, on-board functions, with the exception of redundancies (hot redundancy if any may need to be implemented); EM quality would be commercial components from the same manufacturer of the flight hi-rel components.

In case NASA request such a model as deliverable to them, this EM would become an additional model as ESA would still need a ground test model (currently the AVM) in support of the PFM AIT and of the flight mission.

38.4.1.3 Orbiter PFM

The flight Orbiter S/C, on which all acceptance tests will be performed, but environmental test will have to be executed at qualification levels (keeping the acceptance durations), i.e. sine vibration test, acoustic vibration test, thermal vacuum and thermal balance, EMC RE-RS (radiated emission and susceptibility).

In the case of a mission where ESA develops the Orbiter and a Probe would be part of the system, the Planetary Probe shall be already integrated at the beginning of the PFM Orbiter environmental test campaign.

38.4.1.4 Option: Orbiter STM

An Element level STM (structural and thermal model) is not currently requested, but it could be accommodated within the margins of the presented schedule, if requested by NASA as deliverable model to System AIT, and as technical risk reduction method.

38.4.2 Planetary Probe

Due to the specific characteristic of this product, a risk reduction model philosophy is specified, including a STM, an AVM and the PFM.

38.4.2.1 Planetary Probe STM

The Planetary Probe STM shall be fully representative of the flight structure and thermal control of the flight probe, including external and internal interfaces. It will be subjected to a full qualification test campaign including sine, acoustic vibration and shock, thermal balance and all supplementary tests as required like e.g. alignment and leakage if required. It could be used on the Orbiter STM, in case the latter is built up.

38.4.2.2 Planetary Probe AVM

As per Orbiter (38.4.1.1), in addition it will be integrated to the (NASA) Orbiter AVM for Integrated Orbiter functional tests. Interfaces, power compatibility and data exchange with the Orbiter will be verified there.

38.4.2.3 Planetary Probe FM

The flight probe will be acceptance tested (including environmental acceptance), then delivered to the Orbiter AIT site for integration and joint environmental testing.

38.4.3 Instruments

The instruments of both Orbiter and Probe shall comply with the model philosophy of their carriers. In specific case, and upon Instrument Consortium decision, it may be necessary to procure unit level Qualification Models (QM) or Engineering Qualification Models (EQM) to achieve full performance and environmental qualification by test on selected units (in general all newly designed units, or with substantial design modification from off-the-shelf units). QM units are identical in all parts to their corresponding flight units, while EQM units are the same as QM and FM, but with lower quality EEE components, see 38.4.1.2).

38.5 Development Approach

The proposed development approach makes reference to the following assumptions: TRL = 6 for all products and components by the SRR, and a Mission Adoption by early 2023, with about 6 months time to place the contracts to the product suppliers. Particular attention shall be placed in the contract assignation phase, as the critical products (schedule and TRL critical) should get priority. A special mention to the parachute procurement, as the past heritage is not in favour of smooth procurement; the selection of the Subcontractor for this product should be anticipated if possible.

The ESA Program logic needs to be properly phased with the NASA system development logic. In theory, the Program Milestones on ESA and NASA side shall be consistent with a logic where the System Level (NASA) PDR is first performed, with Element level (ESA) PDR following, then all Subcontractors PDR's . CDR logic shall be with Subcontractors CDR's first, followed by Element level (ESA) CDR, concluding then with the System (NASA) CDR. This sequence of events can be jeopardised by the different way ESA and NASA confirm their projects are financed. This difference may be substantial, impacting ESA funding efficiency and, as a consequence, schedule organisation with milestones logic as well. The timely release of a given budget to Industry is of outmost importance for the proper progress of the project development. Besides, adapting to the pace of a project with another International Organisation (i.e. NASA in this case) leading the master schedule and the system interfaces, may be challenging. As a minimum, a larger management reserve should be considered for this specific kind of projects. Greater schedule margins should also be taken, but that is more complicated, being possible only by using consolidated design and trying to anticipate the procurement of time or design critical products. This practice would anyway increase the technical risk, where the schedule risk is more unpredictable, as the external interface definition is in the hands of NASA, leaving ESA more exposed than usual to the risks deriving from late changes of the (NASA) system design.

Some development guidelines can be proposed though, in order to minimise the programmatic risks (details in the risk chapter) that are associated to a joint NASA/ESA Project.

38.5.1 Orbiter Development

An aggressive approach is proposed, with Avionics Verification Model (AVM) and Protoflight Model. This approach is made possible by the adequate heritage and experience the European Industry have acquired w.r.t. the planetary orbiter S/C, i.e. Mars Express, Venus Express, Trace Gas Orbiter (Mars orbiter), JUICE (Jupiter Orbiter and planetary probe, under development).

The Orbiter PFM can be independently tested from the rest of the NASA System, until its completion of environmental test campaign. After that, it will have to be delivered to NASA for their completion of the system Level AIT in US.

There are some exceptions to the above, as follows:

38.5.1.1 Additional STM

The need for the procurement and testing of an Orbiter STM is subject to agreements with NASA. It is currently assumed that NASA does not require such a model for their integrated composite testing in US. However, the currently proposed schedule shows a proper margin of time to accommodate the procurement and AIT of such a model. That may become an ESA project’s choice in order to reduce the technical risk associated to the definition of the mechanical and thermal design of the Orbiter.

38.5.1.2 Electrical Interface Testing

The external interfaces between NASA S/C System and the Orbiter will be verified by test. The electrical interfaces will be verified in EU by the provision by NASA of an Electrical Interface Simulator (NEIS). NASA will have to verify their side of the interface with the Orbiter, therefore ESA shall provide an Orbiter EIS (OEIS) representing Orbiter power and data characteristics. A proper way to verify the mechanical interfaces shall be investigated in the next phase.

38.5.1.3 Interface Functional Test

A reduced configuration of the AVM may be delivered to NASA for a limited time, in order to test the Orbiter to system S/C integrated functions and electrical performances.

The way to do this and how to accommodate this activity into the schedule is not described in this study.

38.5.1.4 RTG and RHU

The nuclear equipment planned to be part of the Ice Giants project design, is currently also included into the design of the ESA elements. The assumption here taken is that no activity with active nuclear devices will be performed in Europe. The RTG's and RHU's will be replaced by simulators (i.e. mechanical and thermal simulators) able to represent the relevant properties during environmental testing, excluding the nuclear radiation.

It is recommended that the simulators of these devices are NASA furnished equipment. As an alternative, NASA my need to provide the ICD's of them, for European Industry procurement.

The integration of flight RTG's and RHU's in US will be performed by specialised NASA personnel.

38.5.2 Planetary Probe Development

There is some heritage in Europe for the development of planetary probes, the most successful being Huygens (Titan Lander) in the frame of the joint ESA/NASA Cassini-Huygens mission to the moons of Saturn. The last lander was Schiaparelli, ExoMars 2016 Mission, that despite ill-fated was able to procure a massive amount of positive data, useful in the context of the design of this new program.

To begin with, also the development of the Probe can be decoupled from that of the Orbiter. In this case, decoupling is possible until the beginning of the Orbiter level environmental test campaign. In case of later availability of the Probe, it could be replaced by its STM until the beginning of the last orbiter Integrated System Test. In this case though, the technical risk would increase, due to lack of integrated mechanical, thermal and EMC test.

It is necessary to introduce now the Probe Model Philosophy. It is described in para. 38.4.2 and sub-paragraphs. It is inherited from the ExoMars 2016 model philosophy, with some adaptation to the current context.

It is important here to highlight the criticalities of the development, mostly lying on the parachute System, the Thermal Protection Subsystem, TPS (and the Heat Shield) and the radiation environment.

Depending on the project choices, and due to Jupiter flyby, the radiation environment is critical and affects the design of the ESA elements. This aspect is described in the relevant design chapters of this report, and definitely affect the procurement, the lead time of the components, and therefore the procurement schedule. This subject is not explicitly shown in the schedule as it is assumed that at the time of Mission Adoption a procurement strategy has been defined, based on the project mission and design choices.

The Parachute System is a critical development that may lead to a need for late integration, jeopardising the completeness of the environmental test campaign of the Planetary Probe. The schedule and programmatic risk here is deemed high, as well as the technical risk if the Probe cannot be tested in a complete configuration. The design of the Parachute system would be custom for this mission, it needs specific wind tunnel tests (possibly for 1:1 scale testing, with few facilities in the world able to accommodate such tests) and airborne drop test of complex preparation; it is recommended to anticipate the procurement of this product. This implies an early effort into the design, with a risk related to the low maturity of the system level design.

The Heat Shield and its TPS can be designed, manufactured and tested in Europe. It is recommended to avoid building expensive high power thermal flux test facility allowing 1:1 scale testing, as the development and cost effort is not deemed worth the anyway questionable results that can be obtained. It is assumed in the context of this study that the verification of the TPS against the entry heat flux can be achieved by analysis, supported by sample level testing on the TPS material samples, tests that can be performed in Europe, with a number of suitable test facilities being available. Note that in the proposed schedule in the following chapters, the TPS and heat shield procurement are not highlighted.

38.5.3 Test Matrix

To provide a first insight on the test activities that are required to qualify and accept the orbiter and optionally the Planetary Probe, the following test matrices are proposed. Testing shall anyway be planned according to the requirements of ECSS-E-10-03C European Test Standard, that would probably need tailoring to the Ice Giants project.

38.5.3.1 Orbiter Test matrix

Tests/ Models	AVM	PFM	Satellite I/F Sim (**)	Dummy Satellite (*)	Remarks
Mass properties		Mass, CoG, MoI		Mass, CoG	(*) Dummy Sat for NASA composite structural tests
Functional test	ISST, IST, SVT	ISST, IST, SVT	Electrical I/F test with NASA		(**) Satellite I/F Simulator for NASA composite electrical I/F tests
EMC	CE - CS	CE – CS, RE - RS			Protoflight test on PFM
Sine vibration		X			Protoflight test (notched)
Acoustic		X			Protoflight test
Pressure, alignment		X			
Thermal Vacuum		X			Protoflight test
Thermal Balance		X			Protoflight test
I/F Tests with NASA	Functional I/F with composite S/C	With NASA I/F Sim. (1)	No RF Suitcase TBC		(1) NASA provided Electrical I/F Simulator Power quality test with NASA Power system

Table 38-5: Orbiter test matrix

38.5.3.2 Planetary Probe Test Matrix

Tests/ Models	AVM	STM	PFM	Dummy Satellite TBC (*)	Remarks
Mass properties		Mass, CoG, MoI	Mass, CoG, MoI	Mass, CoG	(*) Dummy Probe for Orbiter structural tests (optional)
Functional test	ISST, IST, SVT		ISST, IST, SVT		
EMC	CE - CS		CE – CS, RE - RS		Protoflight test on PFM
Sine vibration		X			Qualification test (notched)
Acoustic		X	X		
Pressure, alignment		X	X		
Thermal Vacuum			X		Protoflight test
Thermal Balance		X	Reduced		
I/F Tests with Orbiter	Functional I/F with Orbiter S/C		Functional I/F with Orbiter S/C		Orbiter provided Electrical I/F Simulator

Table 38-6: Planetary probe test matrix

38.5.3.3 Special Case: Parachute System and Aeroshell Test Matrix

Note: the detail of equipment level test is not provided. Ref.: ExoMars 2016

	Parachute System		Aeroshell		Remarks
Tests/ Models	QM	FM	QM	FM	
Drop test	X				
Wind tunnel	X		Scaled		
Deployment	PDD	PDD	Shock separation	Shock separation	
Thermal Vacuum	X	X	X		
Vibration	X	X	X	On Probe PFM	
Heat flux			Sample level		No full scale test
Mass Properties	X	X	X	X	

PDD: Parachute Deployment Device

Table 38-7: Special case and parachute system and aeroshell test matrix

38.6 Schedule

For ease of reading, the schedule is split into two parts, the first showing the ESA Orbiter and NASA System AIT, the second showing the Planetary Probe.

38.6.1 Orbiter Schedule

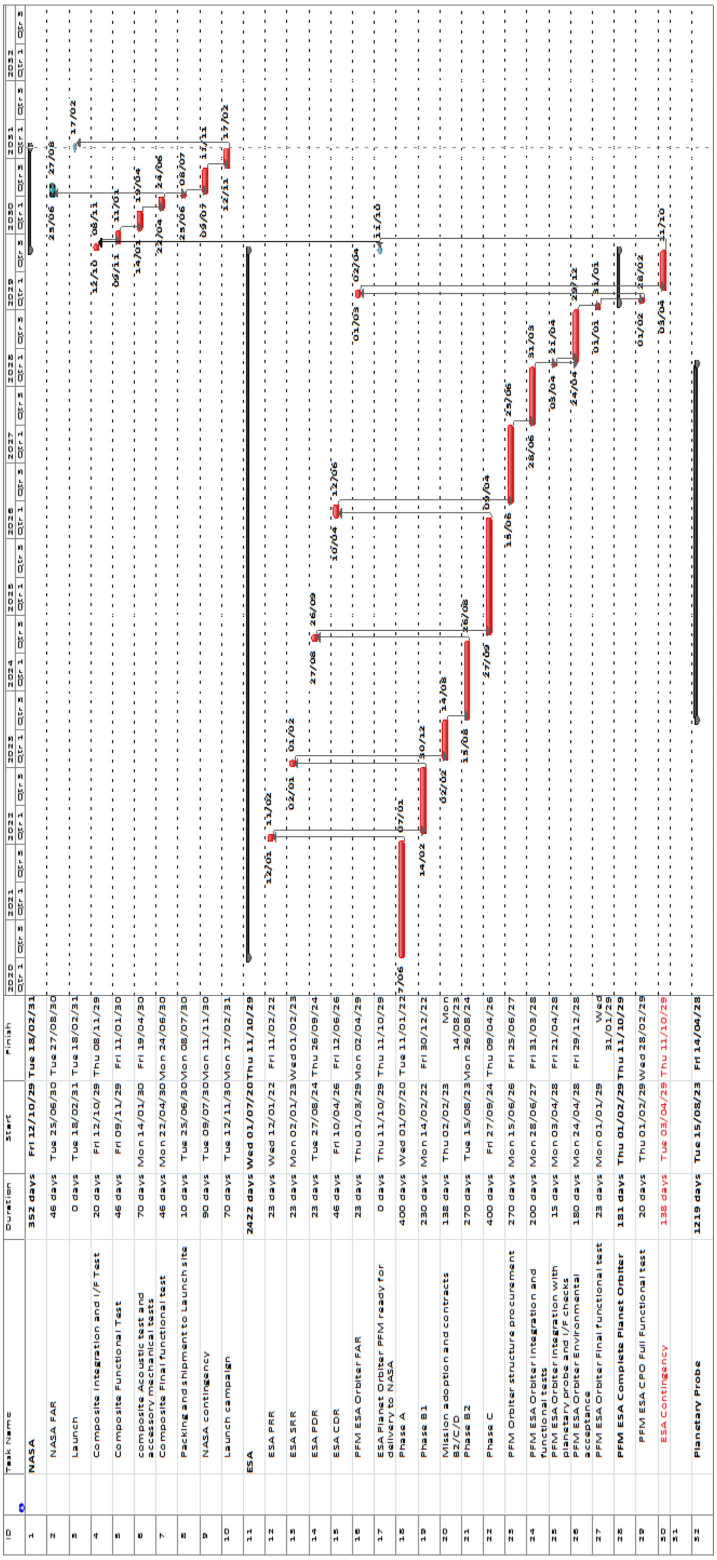

Figure 38-1: Orbiter Schedule

38.6.2 Planetary Probe Schedule

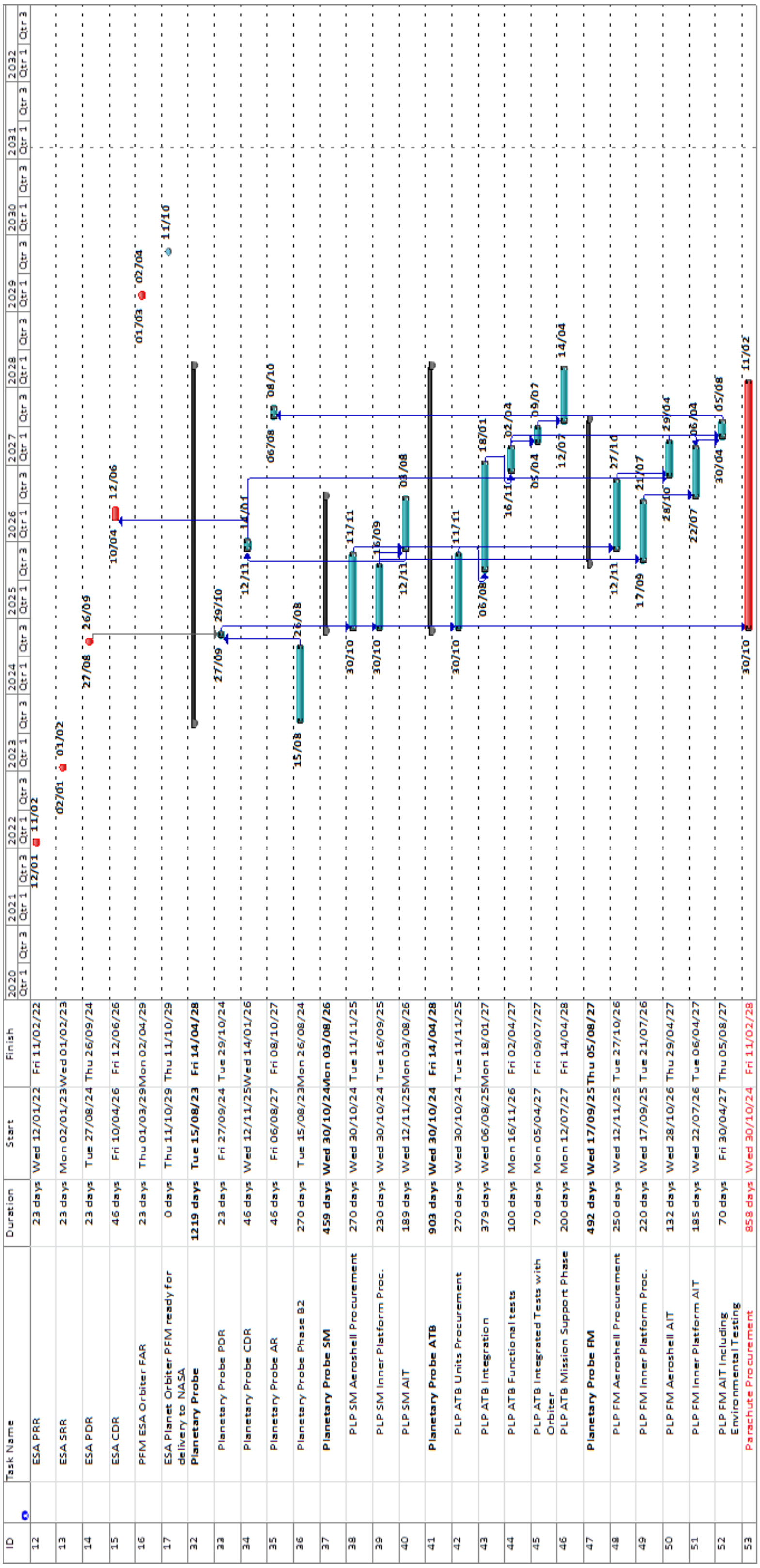

Figure 38-2: Planetary Probe Schedule

38.6.3 Back-up Data: ExoMars 2016 Schiaparelli Probe

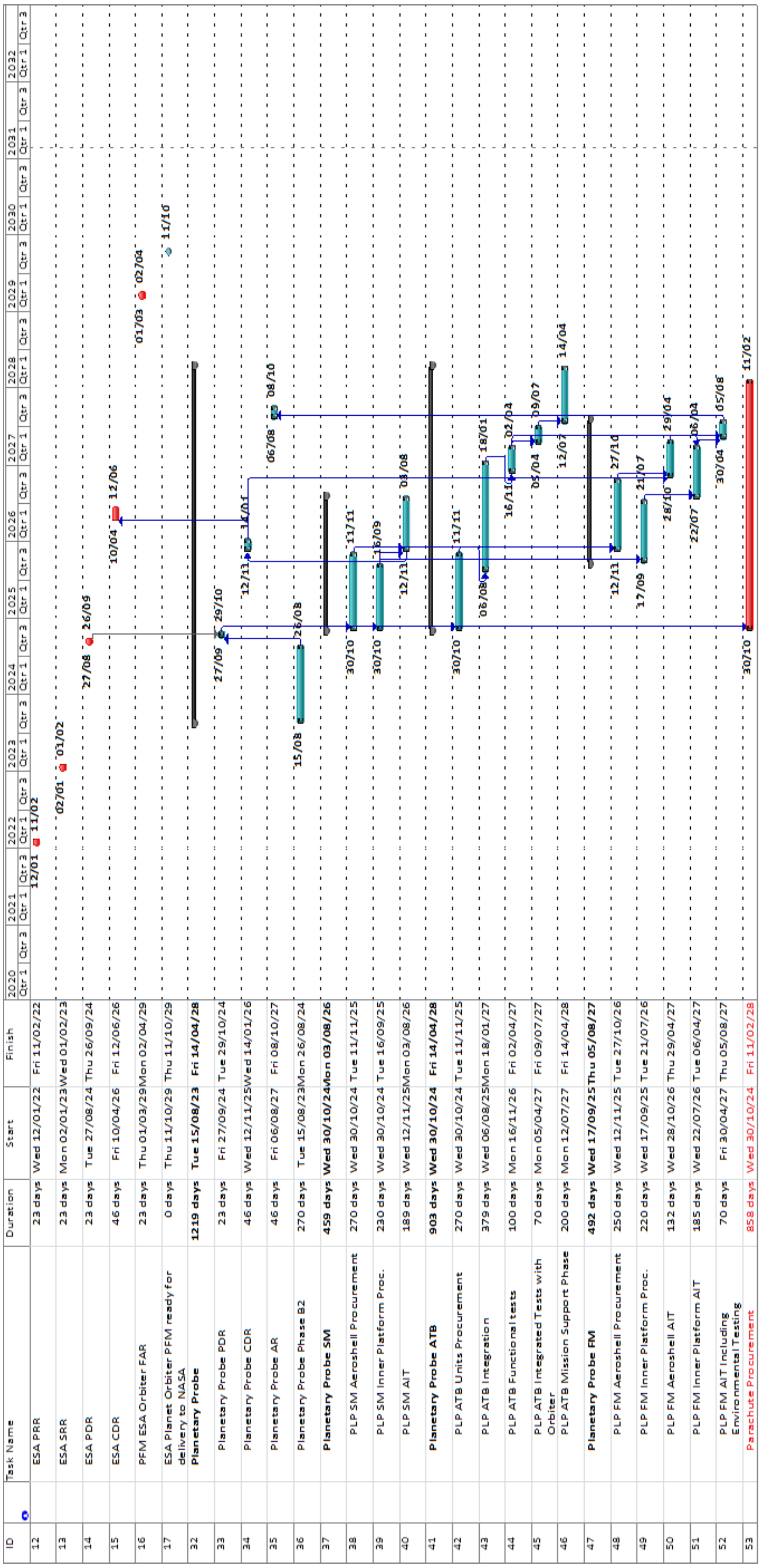

Figure 38-3: Back-up Data Schedule

38.7 Summary and Recommendations

In summary, the proposed approach is compliant with the programmatic requirements.

It is recommended that NASA provides a definition of System Model Philosophy in order for ESA to begin with a negotiation based on their current proposal for the orbiter.

It is important to assess the real scope of the Program on ESA side, in order to cope with the main challenges, that are mostly on the management of the programmatic differences between ESA and NASA on the funding and milestones logic, and on the different ways to activate their Industry.

On ESA side, technical criticalities are on the radiation environment for their elements, and on the Parachute procurement for a Planetary Probe. The TPS is a special procurement to be taken under special observation.

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39 COST

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40 CONCLUSIONS

40.1 Satisfaction of Study Objectives

M* Ice Giants Objectives have been successfully achieved.

Study Objective	Achievement (Y/N)	Remarks
1. Assess the possible European contribution to a NASA-led mission to the Ice Giants with launch in 2029-2034.	**Y**	Study Options identification and down-selection explored diverse architectures for cooperation, in line with the specified timeline and scientific objectives
2. Establish conceptual design for the key European element(s) in order to identify the required resources and define the interfaces with the international partner.	**Y**	Addressed by requirements formulation / design / sensitivity analysis / trade-offs
3. Highlight the technological areas for which mission-enabling developments would be required.	**Y**	Addressed by requirements formulation / design / sensitivity analysis / trade-offs / gaps identification / technology readiness assessment
4. Define the programmatic approach and the schedule constraints for the studied option(s).	**Y**	M* Ice Giants Team included Programmatics expert and specialists with extensive experience in providing required inputs (heritage /TRLs)
5. Assess the mission cost for the studied option(s), taking into account that the ESA contribution shall fit within an M-class mission budget, i.e. 550 MEuro (excluding Member State contributions such as payloads).	**Y**	M* Ice Giants Team included Cost expert and specialists with extensive experience in providing required inputs (heritage /TRLs)

40.2 Probe

40.2.1 Major Findings

The PEP CDF Study was taken as a reference for the probe assessment: only deltas with respect to PEP have been assessed in the M* Ice Giants CDF Study *(Note: M* is ~ 345 kg and PEP was ~ 315 kg).*

Relevant findings are hereafter summarised.

40.2.1.1 Mass

Despite the lower entry velocity (inertial entry velocity was estimated to be ~23 km/s at the beginning of the study, while in PEP it was 24.7 km/s, although this was later revised for M* Ice Giants to 25 km/s) with respect to PEP, the expected mass savings for the TPS did not materialise, as current available knowledge about TPS materials invalidated the optimistic estimates assumed in PEP (2010 timeframe).

In particular, the front shield thickness and mass increased by ~50% compared to the PEP baseline (PEP Front Shield Mass: 73 kg – M* Ice Giants FS Mass: 107 kg)

In addition, the atmospheric pressure range for operations identified as of scientific interest for M* Ice Giants (between 1 and 10 bars) was significantly less than the 100 bars upper limit for PEP, resulting in:

- A lighter pressure vessel structure (~30 kg less than PEP)
- A bigger parachute (as the M* Ice Giants descent timeline foresees 90 mins from 1 to 10 bars, while in PEP the PAS would only sustain the probe descent from 1 to 10 bar in 30 mins, followed by a free fall from 10 to 100 bars in 60 mins – hence requiring significantly less deceleration). As such the M* Ice Giants baseline includes 2 subsonic PAS with ~ 10 kg increase in mass compared to PEP.

The mass increase of the TPS and PAS implied a higher ballistic coefficient, leading to higher heat fluxes and loads, and thus a further increase in the TPS thickness.

In order to limit the ballistic coefficient increase (avoiding further heat flux and load increases), and to accommodate the 2 parachutes, a 10 cm probe diameter increase was baselined. This decision then had a second order mass increase on the probe structure and TPS, which increased along with the increased geometry.

A mass reduction of ~10 kg with respect to PEP for the DHS was achieved via the latest technology developments in the DHS field.

40.2.1.2 RHUs

Given a change to the operations concept such that only timers are switched ON during the coast phase, and no thermal boost before separation from the orbiter, 31 RHUs are considered necessary to survive the 20-day coast phase. These were not considered during the PEP design.

40.2.2 Open Points

The following open points have been identified at the end of the M* Ice Giants study:

- A further consolidation of heat fluxes and loads should be performed, injecting also understanding of the margin policy adopted by NASA partners, for alignment between parties.
- As a result of the heat fluxes/loads updates, further iterations on the TPS should be performed (including the subsequent impacts on PAS/accommodation/structures and ballistic coefficient), until an optimal convergence is obtained.
- A material trade-off for the TPS should also be performed between:

    - Classical fully dense carbon-phenolic ablators (similar to NASA FM5055 used for the Galileo probe, at higher loads compared to the ones computed in M*). These are available in Europe with large annual production rates as they are supplied for nozzles on various launchers and missiles.
    - C-C ceramic materials – characterised by high maturity and flight heritage, however they have never been tested to the heat loads expected for Neptune and Uranus entry.
- Independently from the TPS material considered, no facility is currently available worldwide, which could provide representative test conditions close to those predicted for the mission. For the Jupiter Galileo probe a very large plasma wind tunnel facility was specifically built; since then, there was no other use for the facility, and it was dismantled. Absence of adequate test facilities implies that only testing at sample level could be performed, requiring high margins to be taken in the design.
- The dynamic pressure impact on the structural mass should be duly assessed, as during the M* Ice Giants Study time constraints did not allow for a further iteration of the structures subsystem in relation to this aspect.
- The probe aerodynamic stability must be ensured by design, and it was noted that the final baselined M* Ice Giants probe design would not achieve this. This was due to final changes to the configuration and mass parameters during the IFP with no time to make the necessary further analysis and amendments. In any case, with the other above-listed optimisations and open points still to be addressed, it is considered wise to leave this issue as future work.

40.2.3 Areas for Further Investigation

A dedicated further Probe study is highly recommended, as the CDF Study highlighted that the requirements and constraints for M* Ice Giants are such that the probe cannot just be considered a delta-design with respect to PEP, as it was originally assumed during the Study preparation phase.

In particular, a dedicated study should focus on the following issues:

- Payload operational timeline and scientific requirements shall be clearly identified, as both the required atmospheric pressure level and scientific measurements duration have an impact on the design.
- The impacts of the 50 g deceleration on the instruments shall be checked.
- Radiation effects on the payload shall be assessed.
- The impact of the communications black-out during entry and descent (0 deg elevation for 10 minutes) shall be addressed.
- The estimated DHS mass of 1 kg shall be revisited (considered too optimistic)
- The pressure vessel (DM) structural mass shall be computed via a more detailed analysis.
- The pressure load on the FS shall be analysed.
- The need of a hot structure for the FS shall be reassessed (no significant benefit is recognised at -35 deg FPA).

- The payload mass could be revisited, in order to target an increase (baseline design includes a 11.10 kg payload, however the scientific community has identified a ~20 kg class payload suite which represents a more scientifically meaningful case):
  - By reducing the data rate (e.g. by 50%) the power demand decreases linearly together with the mass of the batteries (as 73% of the energy of the probe is dedicated to communications). This mass saving from the batteries could be re-allocated to the payload;

    Note: With 38,000 km orbiter-probe range and at 80 W RF Power TWTA = 10.8 Mb data is generated in 90 mins at 2 kbps (a significantly larger data volume than for the Galileo Probe by a factor ~3, which considered 1 kbps for ca. 60 mins -> total 3.6 Mb)
  - By reducing the distance between the orbiter and probe (at the cost of some delta-V) the communications power (and thus battery mass) would also decrease and this mass could be allocated to the payload;
  - By reducing the duration of the descent operations (e.g. from 90 mins to 60 mins), both parachute and battery mass could be saved in favour of the payload.
- A FPA = -18 deg could be considered:
  - Note that -35 deg was retained for the M* Ice Giants baseline because of the requirement for Direct-To-Earth (DTE) link visibility during entry. This also allowed for greater reuse of the PEP analysis and design (same FPA), which had been recommended at the beginning of the Study;
  - -18 deg would allow usage of European test facilities for the TPS (for sample level testing)
  - The overall TPS mass impact must still be traded-off (lower heat flux, higher heat loads)
- For the Uranus case, the M* Ice Giants study has assumed identical entry conditions and atmosphere as for Neptune case: a dedicated study shall take into account peculiarities and address ad-hoc design solutions.
- The datahandling subsystem design needs to be revisited and flown down to all the other subsystems due to a design change after IFP.

40.3 Orbiter

40.3.1 Major Findings

Relevant findings concerning the Orbiter design are hereafter summarised:

- The scientific requirement of downloading 32 Gb data volume at the end of the mission (assumed 50 days orbit) is challenging for the Neptune case:
  - In order to meet the requirements, an extended mission duration, on top of the nominal 2 years mission lifetime would be required (see Systems Chapter for details). However:
    - RTGs EOL power still to be confirmed

  - Energy excess in longer considered orbits (e.g. 100 days and 75 days) could be used to modify the timeline and perform more science when more energy is available;
  - The orbits were modelled such that there were 50 /75 /100 days periods, however in reality there will be a gradual reduction across the periods, allowing for more data to be downloaded;
  - The actual orbits used for science operations may also look very different to the 50 / 75 / 100 days profiles considered. This would require further iterations between mission analysis, the system and the payload teams.
  - An upgrade on the ground segment could compensate for the limited resources available on board (current baseline is an array of 2 G/S's, however this could be increased further);
  - The possibility might exist for using the NASA Deep Space Network to augment the ESA ground station network.
- For the Uranus case the data volume downloadable is 53 Gb (with the same communications design as for Neptune).
- 3 RTGs are essential to enable any type of useful science, even for the Uranus case.
- In a dual orbiter scenario the need for 3 RTGs would imply a high number of RTGs (6+), with the following issues to be considered:
  - RTG production rate (~8 years required for 3 eMMRTGs)
  - Launch and launch site safety requirements
  - Ground facilities availability for integration
  - Feasibility of integration through fairing doors
- Trajectory to target Uranus and – in particular – flyby at Jupiter might impose stringent requirements for radiation tolerance of up to 155 krad for sensitive equipment (with shielding of 4 mm Al). There is also much uncertainty in the estimation of the radiation environment which requires further work.
- Technology developments shall be compatible with the programmatic requirement of TRL 6 by end of 2022 (corresponding milestone: mission adoption) for launch on 13 February 2031.

40.3.2 Open Points

The following open points have been identified at the end of the M* Ice Giants study:

- The payload timeline should be consolidated, tailoring it to balance the science objectives with the limits of available resources (particularly power/data). An example could be performing less science at planetary periapsis during the shorter orbits. Such orbits were expected to occur at the end of the mission, by which point the planetary science should already be well characterised. This allows focussing the shorter orbits (which are severely energy constrained) on the Moon flyby science.
- The instruments duty cycles could be reconsidered (e.g. whether the USO must be ON at all times). Due to the energy constraints, any device always ON (even at low power) constrains the operation of other instruments or communications.

- The RTG characteristics should be better understood:
  - Power degradation characteristics (especially EOL power output)
  - Number of RTGs that could actually be made available to an ESA mission element
- Dual orbiter accommodation inside the launcher vehicle trade-off shall be performed.
- Minimal distance between the 2 main engines of the Neptune Orbiter (as per the current baseline) should be checked for thermal and attitude control (e.g. misalignment) implications. The alternative of embarking a single main engine could be a solution (as for the Uranus design case), although this would add additional propellant mass due to gravity losses. Note that a similar issue was faced for ExoMars, and the lessons learned from this could be incorporated.
- The HGA diameter (assumed in the current baseline as 3m) could be reassessed, as the assumed dual launch on an SLS launcher (stacked configuration) would seem to allow enlarging this. In addition, an antenna pointing capability (not baselined in the current design) could also be assessed.
- The structural mass of the design (~7% of the wet mass) is considered optimistic and should be revisited.

40.3.3 Areas for Further Investigation

- Alternative Jupiter flybys for targeting Uranus via different launch dates should be carefully investigated, should the radiation tolerance of equipment prove insufficient. The sensitivity of the radiation environment to launch date should also be better understood. A detailed assessment of the radiation environment must be performed as large uncertainties currently exist, in order to determine requirements for shielding mass and any required technology developments / qualification.
- Alternative radiation shielding materials could be traded off to target mass savings.
- Alternative science tours should be investigated, along with the implications on the mission timeline and scientific data return.
- Energy excess available during cruise (in particular at Jupiter) could be considered for science usage.
- Dedicated design for a specific planet could bring optimization to some of the subsystems (e.g. smaller propellant tanks, structures, reduced communications powers, etc).
  - *Note: due to limited amount of time in the frame of the CDF Study, the orbiter design has been kept identical for both planets. This approach offers an advantage in case one party develops the two orbiters (recurrent).*
- The implications of the launch profile and configuration should be assessed, including requirements for late access under the fairing (e.g. RTGs, RTUs), coupled loads, available volume for antennae and other large structures. In addition, the profile of the cruise (up until separation of the ESA orbiter from the NASA orbiter) should be discussed with NASA to harmonise assumptions.

- The datahandling subsystem design needs to be revisited and flown down to all the other subsystems due to a design change after IFP.
-

40.4 Lander

40.4.1 Requirements and Assumptions

One design session of the CDF M* Ice Giants Study was dedicated to perform a high-level feasibility assessment for a Lander at Triton, Neptune's largest Moon. This was envisioned to deliver a payload of 11.18 kg to the surface.

The main requirements considered were as follows:

- The Triton Lander shall be released from Triton flyby.
- The Triton Lander shall perform a soft landing manoeuvre.
- The Triton Lander shall operate during one week of lifetime.

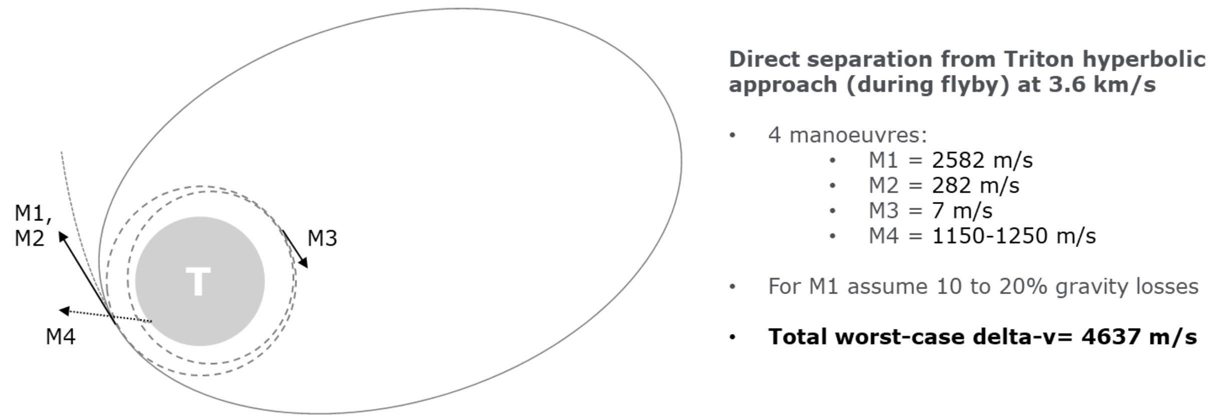

Figure 40-1: Lander release from Triton flyby

It was assumed that there is no atmospheric contribution to braking (or heating), and the worst-case delta-v for performing a soft landing would be 4,637 m/s (see Figure 40-1).

40.4.2 Major Findings

A rough scaling exercise from an existing lander study was performed, in order to derive a quick relationship as displayed in the graph below. Note that this does not include the impacts of propellant tank and surrounding structure sizing as a function of wet mass, and as such is considered optimistic at higher wet masses.

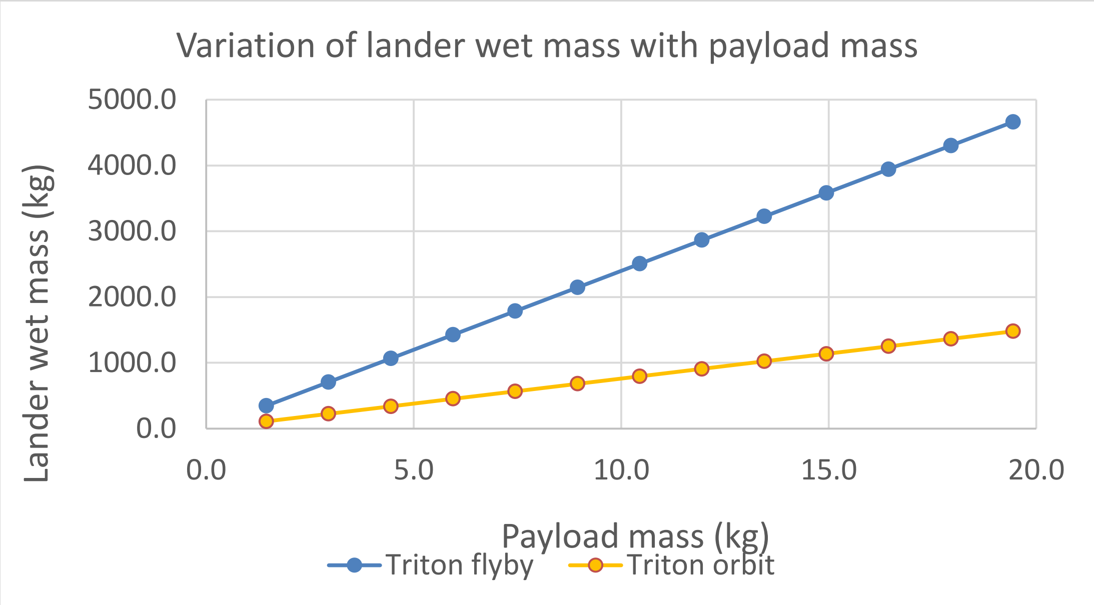

Figure 40-2: Variation of Lander wet mass with payload mass

As can be seen, in order to land ~10 kg payload from Triton flyby, it was calculated that a Lander of wet mass of approximately 2.0-2.5 tons would be required.

40.4.3 Open Points and Areas for Further Investigation

The following open points and areas for further investigation have been identified at the end of the M* Ice Giants session dedicated to the Lander:

- Plume impingement contamination during descent
- Landing gear sizing coping with unknown surface characteristics
- Lander release strategy (from flyby or from orbit) and impact on lander and carrier (orbiter)
- Visibility (range, elevation) of the carrier (orbiter) for communication relay.

A list of main technology developments has been identified:

Subsystem	**Technology**
PROPULSION	Throttled / pulsed propulsion capabilities in a closed-loop GNC system for the final descent manoeuvre
GNC	Dual-use LIDAR, hazard-detection-avoidance and altimeter
POWER	Combined avionics/power unit approach, as “Minavio” concept
OPERATIONS	Array of up to 4 ground stations, with 2 receiver-only antennas, and 2 Transmitting + receiving

40.5 Additional Observations

During the Internal Final Presentation, further discussions with NASA were identified as highly beneficial to harmonise assumptions, exchange design-relevant expertise (e.g. radiation analysis, atmospheric entry) and to discuss potential areas of shared interest such as on ground station usage and RTGs. Further to this, a few additional observations were recorded and are hereafter reported for completeness:

Scientific Objectives

A mission including an Orbiter and a Probe to the same planet would cover most of the scientific objectives, however both planets have unique aspects that are of high interest for the scientific community. A trade-off between addressing only one planet (covering all science objectives) or two planets (addressing less scientific objectives) must be performed as part of future work.

Critical Technical Issues

The following main technical issues were highlighted for the probe:

- PAS procurement should be performed with an approach to minimise risks (based on negative experience from ExoMars, with supplier bankruptcy)
- TPS full testing should be avoided in order to save on cost and schedule. Sample testing and sufficient design margin should rather be adopted.

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41 REFERENCES

41.1.1.1 Chapter 3 Science Objectives References

RD[1] Ice Giants Pre-Decadal Study Final Report, JPL D-100520, June 2017

RD[2] Solar System Ice Giants: Exoplanets in our Backyard, Exoplanet Science Strategy White Paper, submitted to the US National Academies of Sciences, Engineering and Medicine, Space Studies Board, 9 March 2018 [https://arxiv.org/abs/1804.03573]

RD[3] Neptune and Triton: Essential pieces of the Solar System puzzle, A. Masters et al., Planetary and Space Science, 104 (2014), 108–121

RD[4] Outer Solar System Exploration: A Compelling and Unified Dual Mission Decadal Strategy for Exploring Uranus, Neptune, Triton, Dwarf Planets, and Small KBOs and Centaurs, Amy A. Simon et al. white paper submitted, 2018 [https://arxiv.org/abs/1807.08769]

RD[5] Scientific rationale for Uranus and Neptune in situ explorations, Mousis et al., Planetary and Space Science 155 (2018) 12–40

RD[6] The Comparative Exploration of the Ice Giant Planets with Twin Spacecraft: Unveiling the History of our Solar System. Turrini D., et al., Planetary & Space Science 104 (2014), 93-107.

41.1.1.2 Chapter 4 Neptune Mission Analysis References

RD[7] CDF Study Report PEP CDF-106 European Space Agency, July 2010.
PEP (V), Planetary Entry Probe to Venus, CDF-106(A),
PEP (SUN), Planetary Entry Probe to Saturn, Uranus and Neptune, CDF-106(B)

41.1.1.3 Chapter 5 Neptune Systems References

RD[8] Space Launch System (SLS), Mission Planner's Guide, ESD 30000, NASA, Initial Baseline, 12 April 2017

41.1.1.4 Chapter 6 Neptune Payload References

RD[9] Scientific rationale for Uranus and Neptune in situ explorations, Mousis, O., et al., Planetary and Space Science, Volume 155, p. 12-40, 2018

RD[10] The Hera Saturn entry probe mission, Mousis, O., et al., Planetary and Space Science 130, p. 80-103, 2016

RD[11] Scientific rationale for Saturn's in situ exploration. Mousis, O. et al., Planetary and Space Science 104, p. 29-47, 2014

41.1.1.5 Chapter 8 Neptune Structures References

RD[12] Reference Roark's formulas for Stress and Strain, 7th Edition, page 737, Table 15.2, number 22.

RD[13] 2017-04-SLS-MissionPlanersGuide-0005323

41.1.1.6 Chapter 9 Neptune Mechanisms References

RD[14] Huygens separation mechanisms. Dr. Udo R. Herlach, P. Tatalias, B. Schmid, D. Musset.

RD[15] Development of a deployable arm to hold mirrors for space interferometric instrument. Mansoor, Zain.

RD[16] Dynamics and robust adaptive control of a deployable boom for a space probe. ZhongYi Chu, YiAn Lei, Dan Li

41.1.1.7 Chapter 10 Neptune Propulsion References

RD[17] (LEROS 4B - http://www.lolannaicker.com/documents/SP2014_2969298.pdf)

RD[18] (http://www.space-propulsion.com/spacecraft-propulsion/bipropellant-thrusters/10-bipropellant-thrusters.html)

RD[19] (http://epic-src.eu/wp-content/uploads/21_EPICWorkshop2017_ARIANEGROUP_2017-10-25-SMA-Presentation_EPIC.pdf)

RD[20] (https://www.rocket.com/files/aerojet/documents/Capabilities/PDFs/Bipropellant%20Data%20Sheets.pdf)

RD[21] (http://www.space-propulsion.com/spacecraft-propulsion/hydrazine-thrusters/20n-hydrazine-thruster.html)

41.1.1.8 Chapter 11 Neptune AOCS References

RD[22] Star sensing Based Safe Mode, TRP Final Report – ESA contract 4000112606/14/NL/MH, Airbus Defence & Space, 2017

RD[23] Hydra Star Tracker for JUICE Mission, AAS guidance and control conference, Breckenridge, B. Gelin, G. Montay, Y. Henriquel, J-F. Bouvry, P. Regnier and D. Gherardi, 2018

RD[24] Design of a vision-based navigation camera for JUICE, ESA guidance, navigation and control conference, L. Nicollet, C. Chalté, L. Majewski, M. Beaumel, M. Chapuy, K. Kanani, D. Gherardi, 2017

RD[25] Magnetic Fields of the Outer Planets, Space Science, C. Russell, M. Dougherty, 2010

41.1.1.9 Chapter 12 Neptune GNC References

RD[26] Cassini/Huygens Interface Requirements Documents, section 2.4.2 (probe delivery) and section 2.4.3 (release), December 30, 1994

RD[27] Huygens Mission Operations Plan, section 3.3.4 (probe separation), 16 December 2004

41.1.1.10 Chapter 13 Neptune Power References

RD[28] United States Government Accountability Office: Report to Congressional Requestors. GAO-17-673. September 2017.

41.1.1.11 Chapter 14 Neptune Telecommunications References

RD[29] ECSS-E-ST-50C Rev. 2, *Radio Frequency and Modulation*, 4 October 2011, complemented with CCSDS 131.2-B.1, *Flexible advanced coding and modulation scheme for high rate telemetry applications,* March 2012.

RD[30] ECSS-E-50-04C, *Telecommand protocols synchronization and channel coding*, 31 July 2008 complemented with CCSDS 231.B-3, *TC synchronization and channel coding*, September 2017.

RD[31] ECSS-E-50-01C*, Telemetry synchronization and channel coding,* 31 July 2008.

RD[32] ECSS-E-50-02C, *Ranging and Doppler Tracking*, 31 July 2007, complemented with CCSDS 4.1.4-B-2, *Pseudo-Noise (PN) Ranging Systems*.

RD[33] Space frequency coordination group (SFCG), resolutions and recommendations available at www.sfcgonline.org.

41.1.1.12 Chapter 15 Neptune Data Handling References

RD[34] SAVOIR Generic OBC Specification, SAVOIR-GS-001 Is. 1, rev1 10-Feb. 2015

RD[35] SAVOIR Functional Reference Architecture, SAVOIR-TN-001 Is. 1, rev4. 4-Dec.-2015

RD[36] ECSS-E-ST-50-12C

RD[37] ECSS-E-ST-50-13C

RD[38] ECSS-E-ST-50-14C

RD[39] ECSS-E-ST-50-15C

RD[40] Extremely Low Power Timer Board EM for Landers - Summary Report - ELPT.TNËOOI2-RSA, 28 Nov 2013

41.1.1.13 Chapter 16 Neptune Thermal References

RD[41] Thermal Protection System Technology and Facility Needs for Demanding Future Planetary Missions; B. Laub and E. Venkatapathy, NASA Ames Research Center; IPPW-2003

RD[42] Overview of Heatshield for Extreme Entry Environment Technology (HEEET) Project; D. Ellerby et al; IPPW-2018

RD[43] Heatshield for Extreme Entry Environment Technology (HEEET) Development Status; D. Ellerby et al; IPPW-2016

RD[44] Investigation and Testing of High Performance to Mass Value Heatshields for Planetary Entry Systems; R. Roumeas (ESA-ESTEC), T. Pichon and A. Lacombe (both SEP division, SNECMA); ESA Bulletin 94, May 1998

RD[45] Advanced Joining Technologies For Hot Structures, J. Barcena et al., Tecnalia; 8th European TPS&HS workshop, 2016

41.1.1.14 Chapter 17 Neptune Aerothermodynamics References

RD[46] Neptune Aerocapture Systems Analysis, Mary Kae Lockwood, *NASA Langley Research Center, Hampton, Virginia*

RD[47] CDF Study Report CDF-043(A), JEP Planetary Entry Probes to Jupiter , ESAT-ESTEC, Dec 2005

RD[48] J.P. Huot, ESA-ESTEC 2010, internal ESTEC communication

RD[49] Aurora Program Earth reentry demonstrator EVD, 2005

RD[50] Program TRAJ3D Three degree of freedom planetary entry and descent trajectory simulation user manual. Fluid Gravity Engineering LTD

RD[51] Hannah Boehrk , "Zur induktiven Nachheizung einer Überschallwasserstoffströmung " , PhD thesis , 2009

RD[52] S. Loehle, IRS private communication 2019

41.1.1.15 Chapter 18 Neptune EDS Parachute References

RD[53] Software User Manual, Aerodynamic Design Tool for Parachutes and Enhancing Test Facilities for Parachutes. VOR-UM-1301. Contract Number 4000105673/NL/CP.

RD[54] Galileo Parachute System Performance, H. McMenamin, 14th Aerodynamic Decelerator Systems Technology Conference AIAA-97-1510.

41.1.1.16 Chapter 34 Triton Lander References

RD[55] Europa Lander Study. 2016 Report. Europa Lander Mission Pre-Phase A. JPL D-97667. https://europa.nasa.gov/resources/58/europa-lander-study-2016-report/

RD[56] Europa Lander SDT Report & Mission Concept OPAG. Feb 22, 2017, Atlanta, GA.https://www.lpi.usra.edu/opag/meetings/feb2017/presentations/Hand.pdf

RD[57] Landing on Europa: Key Challenges and an Architecture Concept. 15th International Planetary Probe Workshop, Boulder Colorado, June 2018.

RD[58] https://www.colorado.edu/event/ippw2018/sites/default/files/attached-files/outersys_3_zimmer_presid512_presslides_docid1205.pdf

RD[59] Europa Lander Mission Overview and Update. 15th International Planetary Probe Workshop, Boulder Colorado, June 2018

RD[60] https://www.colorado.edu/event/ippw2018/sites/default/files/attached-files/outersys_1_sell_presid716_presslides_docid1199.pdf

RD[61] Proposed Europa Lander Descent Stage Overview. 15th International Planetary Probe Workshop, Boulder Colorado, June 2018

RD[62] https://www.colorado.edu/event/ippw2018/sites/default/files/attached-files/outersys_2_kulkarni_presid642_presslides_docid1204.pdf

RD[63] GNC Design and Performance of the ExoMars EDL Demonstrator. P. Martella et al. 8th International ESA Conference on Guidance, Navigation & Control Systems. 5-10 June 2011.

41.1.1.17 Chapter 35 Radiation References

RD[64] NASA Ice Giants Pre-Decadal Study Final Report, JPL D-100520, June 2017

RD[65] Xapsos, M. A., G. P. Summers, J. L. Barth, E. G. Stassinopoulos, and E. A. Burke, Probabillity Model for Cumulative Solar Proton Event Fluences, IEEE Trans. Nucl. Sci., 47, 486-490, 2000

RD[66] Jiggens, P., Varotsou, A., Truscott, P., Heynderickx, D., Lei, F., Evans, H. & Daly, E., The Solar Accumulated and Peak Proton and Heavy Ion Radiation Environment (SAPPHIRE) Model, IEEE Transactions on Nuclear Science, Volume 65, Issue 2, 2018

RD[67] ECSS-E-ST-10-04C Space Environment

RD[68] Divine N. and Garrett H. B., Charged particle distribution in Jupiter's magnetosphere, J. Geophys. Res., 88, 6889-6903, 1983

RD[69] Garrett H. B. et al, Galileo Interim Radiation Electron Model, Jet Propulsion Laboratory, California Inst. of Technol., JPL 03-006, Pasadena, CA, 2003

RD[70] Jun I., Henry B. Garrett and Robin W. Evans, "High-Energy trapped Particle Environments at Jupiter: an update", IEEE Transactions on Nuclear Science, vol. 52, No. 6, 2005

RD[71] H. B. Garrett, M. Kokorowski, and I. Jun, Galileo Interim Radiation Electron Model Update—2012, JPL Publication 12-9

RD[72] Sicard A. and Bourdarie S., Physical electron belts model from Jupiter's surface to the orbit of Europa, J. Geophys. Res., 109, 2004

RD[73] Bourdarie S. and Sicard A., Jupiter environmental modelling, ONERA technical note 120 issue 1.2, ESA contract 19735/NL/HB, FR 1/11189 DESP, October 2006

RD[74] E. Maddox, B. Taylor, G. Vacanti, Interplanetary Electron Model Tools Software Documentation, Report of ESA contract 22915/09/NL/AF, CSC-IEM-DOC003, 2011

RD[75] P. Truscott, "JORE2M2 Project: SHIELDOSE-2Q Requirements, Interface Control, Design, Implementation and Justification Document," QINETIQ/TS/AS/SDD0900017, 2010

RD[76] S. M. Seltzer, "Updated calculations for routine space-shielding radiation dose estimates: SHIELDOSE2," National Institute of Standards and Technology report NISTIR 5477, December 1994.

RD[77] Hugh D. R. Evans et al., Jovian Radiation Belt Models, Uncertainties and Margins, IEEE Trans Nucl Sci Vol. 60, No. 4, August 2013

RD[78] Q. Nénon, A. Sicard, P. Kollmann, H. B. Garrett, S. P. A. Sauer, C. Paranicas, A Physical Model of the Proton Radiation Belts of Jupiter inside Europa's Orbit, JGR Space Physics, 2018

RD[79] Henry Garrett and Robin Evans, The JPL Neptune Radiation Model (NMOD), JPL Publication 17-2, July 1, 2017

RD[80] Selesnick, R.S., and Stone, E.C., “Energetic electrons at Uranus: Bimodal diffusion in a satellite limited radiation belt,” J. Geophys. Res., vol. 96, no. A4, pp. 5651–5665, 1 April 1991

RD[81] Henry Garrett and Robin Evans, The JPL Uranian Radiation Model (UMOD), JPL Publication 15-7, September 2015

RD[82] B. H. Mauk and N. J. Fox, Electron radiation belts of the solar system, JGR Vol. 115, A12220, doi:10.1029/2010JA015660, 2010

41.1.1.18 Chapter 36 GS&OPS References

RD[83] ECSS-E-70-41C Telemetry and Telecommand Packet Utilization,15 April 2016

RD[84] CCSDS 727.0-B-4, CCSDS File Delivery Protocol, Blue book, January 2007

RD[85] ECSS-E-ST-50C, Communications, 31 July 2008

RD[86] CCSDS 133.0-B-1, Space Packet Protocol, Cor 2. Sep 2012

RD[87] ECSS-E-70-11C, Space Segment Operability, 31 July 2008

RD[88] Future Architecture of ESA Deep Space Stations for Enhanced Mission Support, REP/1139/2579, issue 1.0, 4 March 2011

RD[89] Future Architecture of ESA Deep Space Stations for Enhanced Mission Support, FAST-INSA-5000-FR-01 16 May 2011.

RD[90] Cost Optimisation for Deep Space Receive-Only Antennas, final report, TN-1016048-01200. Release 1.5, 2018-09-14

RD[91] Rosetta end of Mission Lessons Learned public presentation. RO-ESC-MN-0111

41.1.1.19 Chapter 37 Risk References

RD[92] Space Project Management, Risk Management, ECSS-M-ST-80C, 31 July 2008.

41.1.1.20 Chapter 38 Programmatics References

RD[93] TEC-SYC Cost Risk Procedure, TEC-SYC/5/2010/PRO/HJ, February 2010

RD[94] ESA Cost Engineering Charter of Services, Issue 4, TEC-SYC/12/2009/GRE/HJ

RD[95] ECSS-E-AS-11C Adoption Notice of ISO 16290, Space Systems – Definition of the Technology Readiness Levels (TRLs) and their Criteria of Assessment, dated 1 October 2014. To be supersede by EN16603-11.

42 ACRONYMS

Acronym	Definition
AIT/V	Assembly, Integration and Test/Verification
AIV	Acceptance, Integration and Validation
AOCS	Attitude and Orbit Control System
AOGNC	Attitude and Orbit Guidance, Navigation and Control
AOS	Acquisition of Signal
APE	Absolute Performance Error
ARES	Analysis and Reporting System
ASI	Atmospheric Structure Instrument
AVM	Avionics Verification Model
BC	Back Cover
BCR	Battery Charge Regulator
BDR	Battery Discharge Regulator
BoL	Beginning of Life
BSSM	Back Shell Separation Mechanism
C/SiC	Ceramics/Silicium-Carbon
CaC	Cost at Completion
CDF	Concurrent Design Facility
CDMS	Computer & Data Management System
CDMU	Computer & Data Management Unit
CDR	Critical Design Review
CF	Confluence Fitting
CFDP	CCSDS File Delivery Protocol
CFRP	Carbon Fibre Reinforced Plastics
CMA	Cost Model Accuracy
CoG	Centre of Gravity
COPS	COmetary Pressure Sensor
CTP	Science Core Technology Programme
DDOR	Delta-Differential One-Way Ranging
DDS	Data distribution system

Acronym	Definition
DFMS	Double Focusing Mass Spectrometer
DHS	Data Handling System
DM	Descent Module
DMM	Design Maturity Margin
DOA	Degree of Adequacy of the cost model
DOD	Depth Of Discharge (Battery)
DOE	(US) Department Of Energy
DOR	Differential One-Way Ranging
DSN	NASA Deep Space Network
EAA	Earth Aspect Angle
ECSS	European Cooperation for Space Standardisation (Standards)
EDL	Entry, Descent and Landing
EDS	Entry Descent System
EGOS-CC	ESA Ground Operations System – Common Core
EIP	Entry Interface Point
EM	Engineering Model
EMC	Electro-Magnetic Compatability
eMMRTG	Enhanced Multi-Mission Radioisotope Thermoelectric Generators
EODL	End Of Design Life
EOL	End of Life
EOM	End Of Mission
EPE	External Project Events
EPS	Electrical Power Subsystem
EQM	Engineering and Qualification Model
ESI	European Standard Initiator
ESOC	European Space Operations Centre
ESP	Emission of Solar Protons
FCT	Flight Control Team
FDIR	Failure Detection Isolation and Recovery
FEA	Finite Element Analysis
FEM	Finite Element Model

Acronym	Definition
FM	Flight Model
FoV	Field of View
FPA	Flight Path Angle
FS	Front Shield
FSSM	Front Shield Separation Mechanism
Gbits	Giga Bits
GIRE	Galileo interim radiation electron model
GSE	Ground Support Equipment
GSTP	General Support Technology Programme
HGA	High Gain Antenna
HK	HouseKeeping
HKTM	Housekeeping engineering telemetry.
HW	HardWare
ICD	Interface Control Document
IDST	Integrated Deep-Space Transponder
IMU	Inertial Measurement Unit
INL	Idaho National Laboratory
IQM	Inherent Quality of the cost Model
JANUS	Jovis, Amorum ac Natorum Undique Scrutato
JOSE	JOvian Specification Environment
JUICE	Jupiter Icy moons Explorer
KSC	Kennedy Space Center
LGA	Low Gain Antenna
LoS	Line of Sight
LOS	Loss of Signal
LTA	Long term archive
LV	Launch Vehicle
LV	Latch Valve
MAJIS	Moons And Jupiter Imaging Spectrometer
MIPS	Million Instruction Per Second
MLI	Multi-Layer Insulation

Acronym	Definition
MMRTG	Multi-Mission Radioisotope Thermoelectric Generator
MMU	Mass Memory UNIT
MOC	Mission Operations Centre
MoI	Moment of Inertia
MP	Main Parachute
MS	Mass Spectrometer
MTU	Mission Timer Unit
NEIS	NASA Electrical Interface Simulator
NMOD	Neptune Radiation Model
NO/NC	Normally Open/Normally Closed
NOI	Neptune Orbit Insertion
OAA	Orbiter Aspect Angle
OBC	On-Board Computer
OBCP	Onboard Control Procedures
OBRAS	On-Board Radio Science
OCDT	Open Concurrent Design Tool
ODM	Orbit Deflection Manoeuvre
OEIS	Orbiter Electrical Interface Simulator
PAS	Parachute Assembly System
PC	Pilot Chute
PCDU	Power Conditioning And Distribution Unit
PDD	Parachute deployment Device
PDR	Preliminary Design Review
PEP	Planetary Entry Probe
PER	Parachute Engineering Tool
PFM	Proto-Flight Model
PLF	Payload Fairing
POE	Project Owned Events
PV	Pyro Valve
QIV	Quality of the Input Values
QM	Qualification Model

Acronym	Definition
QSL	Quasi-Static Load
RCS	Reaction Control System
RD	Reference Document
REQ	Requirement
RF	Radio Frequency
RFDN	Radio Frequency Distribution Network
RFM	Recurrent Flight Model
RHA	Radiation Hardness Assurance
RHU	Radio-isotope Heater Unit
RIU	Remote Interface Unit
RN	Neptune Radius
RPE	Relative Performance Error
RPS	Radioisotope Power System
RTG	Radioisotope Thermoelectric Generators
RTOF	Reflectron type Time-Of-Flight
RU	Uranus Radius
S/C	Spacecraft
SAVOIR	Space Avionics Open Interface ARchitecture
SEE	Single Event Effect
SEM	Spin and Eject Mechanism
SEP	Solar Electric Propulsion
SEP	Solar Energetic Particle
SHIELDOSE	Computer code for space-shielding radiation dose calculations
SLS	Space Launch System
SOC	State Of Charge (Battery)
SOC	Science Operations Centre
SPE	Solar Particle Event
SPF	Single Point Failure
SRR	Systems Requirements Review
SSMM	Solid State Mass Memory
SSPA	Solid State Power Amplifier

Acronym	Definition
SST	Study Science Team
STM	Science Traceability Matrix
SW	SoftWare
T/M	Thrust-to-Mass
TBC	To Be Confirmed
TBD	To Be Decided
TC	Telecommand
TID	Total Ionizing Dose
TIDL	Total Ionising Dose Level
TIDS	Total Ionising Dose Sensitivity
TM	Telemetry
TMTC	Telemetry & Telecommand
TNID	Total Non Ionising Dose
TPS	Thermal Protection System
TRL	Technology Readiness Level
TT&C	Tracking Telemetry And Command
TWTA	Travelling Wave Tube Amplifier
UMOD	Uranian Radiation Model
UOI	Uranus Orbit Insertion
USO	Ultra-Stable Oscillator

A TRACEABILITY MATRICES

Science Theme	Science Objective	Scientific Measurement Requirements: Physical Parameters	Scientific Measurement Requirements: Observables	Instrument	Instrument Requirements	Mission Requirements	Comment
Origins & Interior	1. Constrain the structure and characteristics of the planet's interior, including layering, locations of convective and stable regions, internal dynamics	Magnetic field geometry and time variability	Magnetic field strength and orientation	Magnetometer		Magnetic cleanliness of spacecraft	
		Gravity moments, J2-J6	Perturbations to s/c orbit (also useful to make astrometric observations of rings and satellites)	Radio science package		Close periapse passess	Radio science package also allows for radio occultation measurements to determine the planetary shape and the atmospheric density profile.
	2. Improve knowledge of the planetary dynamo	In situ magnetic field direction and magnitude	Magnetic field direction and magnitude	3-axis Magnetometers on boom	0.1 to 20,000 nT, 1 second cadence	Multiple close orbits; longitude and latitude coverage for degree and order at least 4, preferably 15	
		Remote sensing of magnetic field footprint.	UV and IR emission from auroral and satellite footprints	IR, UV spectral imager(s)	1600-1800 Ang imaging and 3.4–4 micron imaging.		Should not drive the UV or IR instrument, but should be considered regarding instrument capability and operations
			Auroral radio emission	Radio Receiver, at least 2 axis electric antenna	10 kHz to 1 MHz, direction finding ability	Multiple close orbits, good longitude and latitude coverage	
Atmosphere	3. Determine the planet's atmospheric heat balance	Net thermal emission	Broadband thermal IR emission	Thermal IR bolometer	5–900 cm^-1, accuracy 1%	Full phase angle coverage	Range based on ~1% of peak for Neptune, accuracy based on 0.1% of peak, See Li et al, 2010, Fig. 1 as reference
		Bond albedo	Visible wavelength bond albedo	Photometer, or suitably calibrated imager or spectrometer	0.3–1.6 µm, accuracy 1%	Full phase angle coverage	Range based on ~10% of peak flux of Sun, and similar to Voyager IRIS shortwave radiometer bandpass. Accuracy assumed to reduce error bars to 1% of total. May be too stringent.
	4. Measure planet's tropospheric 3-D flow (zonal, meridional, vertical) including winds, waves, storms and their lifecycles, and deep convective activity	Vertical, zonal, and meridional profiles of wind speed	Repeat maps of cloud tracers at multiple wavelengths including methane bands	Visible/Near-IR imager, 8 channels	30km/pixel spatial resolution, 3 emission angles (nadir,near limb, intermediate), 425+-25nm, 500+-25nm, 619+-5nm, 653+-25nm, 727+-5nm, 750+-5nm, 890+-10nm, 925+-5nm, repeat one rotation later, Absolute I/F's to 5%, SNR:50 for all but SNR:100 for 750nm & 653nm for winds	Global mapping and feature tracking. Each region viewed at 3 emission angles on 2 consecutive rotations.	Radio science package can provide additional and/or complementary information (3D density structure of stratosphere and troposphere, vertical thermal structure of upper atmosphere)
		Distribution of condensible and disequilibrium species.	Hydrogen ortho/para ratio at P ≥ 3 bar	Mid-IR spectrometer	~1,000 km horizontal spatial resolution, mixing ratio ±20%	Global mapping.	Cloud particle size and density distribution was proposed to be added to this objective, but that is a separate, lower-ranked science objective.
			3-D distribution of hydrocarbons, GeH4, AsH_3, PH_3, in the stratosphere	Near to mid-IR spectrometer	~1,000 km horizontal spatial resolution, mixing ratio ±20%	Global mapping.	Probe can give vertical distribution ONLY in one location. Radio science package can provide additional and/or complementary information (see above).
			3-D distribution of CO, PH_3 in stratosphere	Millimeter/submm spectrometer	~1,000 km horizontal spatial resolution, mixing ratio ±20%	Global mapping	
			3-D distribution of CH4 in upper troposphere	Visible/Near-IR spectrometer	~1,000 km horizontal spatial resolution, mixing ratio ±20%	Global mapping	
			3-D distribution of CO, PH_3, NH_3, H_2S, in upper troposphere	Millimeter spectrometer, centimeter radiometer	~1,000 km horizontal spatial resolution, mixing ratio ±20%	Global mapping	
			3-D distribution of NH_3, H_2S, H_2O in deep troposphere	Centimeter to meter-wavelength radiometer	~1,000 km horizontal spatial resolution, mixing ratio ±20%	Global mapping	
Rings	5. Characterize the structures and temporal changes in the rings	Fine-scale radial structure and particle size constraints in selected rings	Ring optical depth in the visible	Imager or photometer (observe stellar occultations)	Imaging with spatial resolution <100 m	Observe rings at multiple longitudes. Repeat > 8 times for spatial variations and ring shapes	
			Ring optical depth at radio wavelengths	Radio science package	High-gain antenna diffraction pattern on rings < 100 m	Spacecraft pass behind the rings as seen from Earth or other flight element at multiple longitudes. Repeat > 3 times for spatial variations and ring shapes	Uplink radio occultations may be useful.
		Spreading rates and other temporal variations	Visible reflectivity	Imager	Spatial resolution < 100 m	Measure selected rings multiple times over months to years	Constrain evolution of longitudinal structures over at least 10 keplerian spreading times. Constrain radial evolution over 10 viscous spreading times
		Dusty ring structure and particle properties	Visible and near-IR reflectivity	Imager	High rejection of off-axis scattered light	Shadow passages for very high phase angle observations.	Search for dust throughout system, from above planet to beyond major satellites. Characterize rings at phase angles >160 degrees
Moons	6. Obtain a complete inventory of small moons,	Location and orbital parameters of small moons,	Sunlight reflected from small moons	Imager (framing camera)	Identify all bodies >0.5 km in radius within 500,000 km of the planet's	Time and data volume to search orbits within 500,000 km of planet center	

Science Theme	Science Objective	Scientific Measurement Requirements: Physical Parameters	Scientific Measurement Requirements: Observables	Instrument	Instrument Requirements	Mission Requirements	Comment
	including embedded source bodies in dusty rings and moons that could sculpt and shepherd dense rings.	constrain their size			center. Repeated observations for orbit determinations.		
	7. Determine surface composition of rings and moons, including organics; search for variations among moons, past and current modification, and evidence of long-term mass exchange / volatile transport	Surface composition and microphysical properties of rings and moons	Surface composition and phase properties	Visible/Near-IR spectral imager	Wavelength: visible to ~5 µm; pixel scale ≤500 m globally, ≤100 m locally/regionally observations at different phase angles [rings and satellites]	Multiple flybys for global mapping of multiple satellites	
		Geologic context on moons for compositional variations	Surface morphology	Visible wavelength imager	Global mapping ≤1 km pixel scale local high-resolution imaging ≤50 m pixel scale, panchromatic as a minimum, stereo views	Multiple flybys for global mapping of multiple satellites, stereo views	
		Search for plumes and identify spatial and temporal variability in activity and dynamics	Scattered sunlight from plumes	Narrow-angle visible imager	≤ 10km/pixel for all satellites, ≤1 km for Triton	High phase angle and terminator imaging, repeated to constrain variability and dynamics	Narrow-angle visible camera can identify plumes with distant observations; NUV sensitivity useful, but panchromatic can be sufficient (and has more total photons for better SNR)
		Constrain the relative abundances of major constituents (e.g., water ice, methane ice, amorphous carbon) on the rings and satellite surfaces	Vis/NIR spectral features	Visible/Near-IR spectral imager	Spatial resolution sufficient to assess compositional trends at level of 1/10th body, and geological context at 1/100th body on all major moon and select small satellites	Flybys of all major moons and select small satellites and distances sufficient to meet spatial resolution requirements (see instrument requirements)	INMS and Mass spectrometer can provide complementary information on surface and/or interior composition by in situ sampling of exospheres and/or plumes during fly-bys (see also goal 10 for Triton).
		Thermal properties of rings and satellite surfaces	Map thermal IR emission as a function of local time-of-day	Thermal mapper (in conjunction with albedo maps from vis/near-IR imaging)	Resolution of a few km	Multiple satellite flybys, inclined orbits for rings	
		Pickup ions in the magnetosphere that originated as neutrals sputtered from satellite surfaces	Measure the composition of pickup ions in the magnetosphere	Plasma and energetic particle detectors	~1 eV to 1 MeV	Multiple, close (~50 km) satellite flybys. Requires supporting measurements from a 3-axis magnetometer on a boom, measuring 0.1 to 20,000 nT, 0.05 second cadence	This is a secondary way to achieve this objective
Moons	8. Map the shape and surface geology of major and minor satellites	Global-scale surface and structural mapping of major satellites and regional mapping of topography and slopes	Visible reflectivity	Imager	~80% surface coverage <0.5-km pixel scale. Monochromatic sufficient.	Multiple satellite flybys, stereo views.	Map the shape and surface geology of major and minor satellites to: understand past and current surface modification processes; constrain crater distributions and relative surface ages; determine topography; provide evidence of subsurface structure; correlate with gravity; search for sources of endogenic heating. NAC can accomplish with observations on approach/departure. Note that a radar sounder for sub-surface structures was considered for this science objective, but was considered secondary due to accommodation issues. Radio science package can allow for similar studies, however.
		Local high-resolution surface mapping, topography, slopes	Visible reflectivity	Narrow-angle visible camera	<100-m pixel scale. Monochromatic sufficient.	Multiple satellite flybys, stereo views	
		Spectral mapping of geologic units	Visible, IR, and UV reflectivity	Spectral imager at visible, IR, UV wavelengths	Visible to ~5 µm, pixel scale 0.5 km; UV pixel scale ≤500 m globally, ≤100 m locally.	Multiple satellite flybys, observations at different phase angles	
		Distribution of mass	Gravity field	Radio science package	Measure J2	Requires multiple satellite flybys, s/c tracking *during* flybys, ~50 km altitude.	Would like to measure k2.
		Thermal properties of surface	Thermal emission, thermal inertia	Thermal mapper	Spatial resolution ~3 km.	Multiple satellite flybys. Also need albedo maps from vis/near-IR imaging	Considered secondary because of concerns about ability to accommodate a thermal mapper.
	9. Determine the density, mass distribution, internal structure of major satellites and, where possible, small	Mass, and mass distribution	Gravitational moments	Radio science package		Multiple, close (~50 km) satellite flybys, tracking during flybys	This objective includes detection of internal oceans and bulk composition. Neptune's irregular satellite Nereid is of particular interest.
		Mass distribution and rheology	Libration, Cassini state, pole position, orbital motion, tides	Narrow-angle visible camera	100 m - 1 km pixel scale imaging	Global imaging at 100 to 1000 m pixel scale, repeated at different orbital true	

Science Theme	Science Objective	Scientific Measurement Requirements: Physical Parameters	Scientific Measurement Requirements: Observables	Instrument	Instrument Requirements	Mission Requirements	Comment
	inner satellites and irregular satellites					anomalies	
		orbit determination	Satellite positions	Narrow-angle visible camera	n/a	frequent distant observations	
		Density	Global shape	Narrow-angle visible camera	100 m - 1 km pixel scale imaging	Global imaging at 100 to 1000 m pixel scale	
				Radio science package		occultations	
		Presence of a conducting layer or dynamo	induced/intrinsic magnetic fields, direction and magnitude	3-axis magnetometers on boom	0.1 to 20,000 nT, 1-s second cadence	Multiple, close (<50 km) satellite flybys	
Triton	10. Determine the composition, density, structure, source, spatial and temporal variability, and dynamics of Triton's atmosphere	Vertical profiles of aerosols, gas and aerosol scattering properties; velocity of atmospheric features	Vis/Near-IR and UV spectral reflectivity and phase function	Spectral imager at visible, IR, and UV wavelengths	Global coverage at km scales, ~100 m resolution in select regions.	Global context and repeat imaging over months to years (minimum would be two observations, Ideally would want to be able to look regularly (every ~1–2 months) over a period of at least 1–2 years) to constrain variability and dynamics; high-phase angle imaging opportunities	
		Atmospheric composition	Neutral and charged particle characterization	Mass spectrometer/INMS	mass range 1 - 64 amu, mass resolution >1,000 M/dM	Multiple Close Flybys, good latitude and LT coverage	Similar measurements can be performed for the exospheric composition of the Uranian satellites.
				Plasma and particle detectors	~1 eV to 1MeV		
				Langmuir probe	sweep ~20 seconds		
				Radio and plasma wave instrument	10's Hz to 100's kHz, 30 second cadence (required cadence may depend on flyby velocity)		
Magnetosphere	11. Investigate solar wind-magnetosphere-ionosphere interactions and constrain plasma transport in the magnetosphere	Measure ion and electron energy spectra vs time	Measure ion and electron energy spectra vs time	Plasma and energetic particle detectors	~1 eV to 1MeV	Multiple Orbits, dayside apogees >25 planetary radii, nightside apogees >50 radii, good LT and Latitude coverage. 54 days upstream (of planetary bowshock) continuous measurements.	
		Measure magnetic field direction and magnitude	Measure magnetic field direction and magnitude	3-axis magnetometer on boom	0.1 to 20,000 nT, 0.05 second cadence	Multiple Orbits, dayside apogees >25 planetary radii, nightside apogees >50 radii, good LT and Latitude coverage. 54 days upstream (of planetary bowshock) continuous measurements	
		Measure electric and magnetic wave fields to determine wave-particle interaction, electron density, boundaries, magnetospheric activity	Measure electric and magnetic wave fields	Plasma wave receiver, 2-axis electric, 3-axis search coil.	few Hz to 100's kHz, 30 second cadence, burst mode ablility	Multiple Orbits, dayside apogees >25 planetary radii, nightside apogees >50 radii, good LT and Latitude coverage. 54 days upstream (of planetary bowshock) continuous measurements.	
		Measure ion composition to separate magnetospheric and solar wind plasma	Measure ion composition	Plasma and energetic particle detectors	~1ev to ~300 keV	Multiple Orbits, dayside apogees >25 planetary radii, nightside apogees >50 radii, good LT and Latitude coverage	
		Measure ENAs	Measure ENAs	ENA imaging instrument (can also measure energetic ion and electrons with the same instrument)	10 to 150 keV/nucleon	Orbits with large apogee distances, good LT and Latitude coverage	

Table A- 1: Orbiter Science Traceability Matrix

Science theme	Science Priority	Science Objective	Scientific Measurement requirements		Instrument	Instrument requirements	Mission requirements	Comment
			Physical Parameters	Observables				
ORIGIN	1	Determine the planet's bulk composition, including abundances and isotopes of heavy elements	Atmospheric composition as a function of depth	He/H_2 abundance	MS HAD	He/H_2 +/- 0.2% At minimum two measurement points (1 and 10 bars)	Measure at P> 1 bar	Level of uncertainty already reached by the HAD aboard the Galileo probe
				Well-mixed abundances of the noble gases Ne/H_2, Ar/H_2, Kr/H_2, Xe/H_2	MS	Ne/H, Ar/H, Kr/H, Xe/H to a precision of ± 10% At minimum two measurement points (1 and 10 bars)	Probe relay. Instrument survival and performance in the extreme P,T environment, especially if deployed to >10 bar.	Probe to 10 bars will not give well-mixed H_2S, H_2O, S and O isotopes, and NH_3 only marginally.
				Vertical profiles of key cosmogenic species C/H, N/H, S/H	ASI MS	Precision ±5% Regular sampling all along the descent		
	2			Isotopic ratios in hydrogen (D/H), helium ($^3He/^4He$), nitrogen ($^{15}N/^{14}N$), carbon ($^{13}C/^{12}C$), oxygen ($^{17}O/^{16}O$ and $^{18}O/^{16}O$), neon and heavy noble gases (He, Ne, Ar, Kr, Xe)	MS	D/H, $^{15}N/^{14}N$ ±5%; $^3He/^4He$ ±3%; $^{17}O/^{16}O$, $^{18}O/^{16}O$, $^{13}C/^{12}C$ ±1%; Ne, Ar, Kr, and Xe isotopes: ±1% At minimum two measurement points (1 and 10 bars)		

Science theme	Science Priority	Science Objective	Scientific Measurement requirements		Instrument	Instrument requirements	Mission requirements	Comment
			Physical Parameters	Observables				
ATMOSPHERE	2	Determine the composition al, thermal and dynamical structure of the atmosphere	Vertical structure of atmospheric temperature and stability	Density, T, P profile, T lapse rate	ASI	Pressure: ± 1% Temperature: ± 1 K from the upper atmosphere to 10 bar.		
	3		Investigate atmospheric winds and wave phenomena	Profile of descent probe telemetry Doppler frequencies	DWE: Probe USO for Doppler tracking	Zonal Winds: ±1 m/s from 0.1-10 bar	Requires USO on spacecraft carrier	
			Vertical mixing and distribution of chemical species	Vertical profiles of NH_3, CH_4, H_2S, H_2O, CO, GeH_4, AsH_3, PH_3, in the stratosphere	ASI MS	+/- 10% Regular sampling all along the descent		CH_4 condensates around 2 bars
			Vertical distribution, structure, composition and properties of clouds and haze layers	Particle optical properties, size distributions, number and mass densities, opacity, shapes, and composition	NEPH	Measurements at two tropospheric pressure levels as 1 bar and 10 bars (at least)	Sampling atmospheric particle, need access to outside air flow; inlet located at the bottom of the probe	
			Radiative energy and thermal balance of the atmosphere	Up & down visible and IR fluxes	Net Flux Radiometer	Up & down visible flux: ~0.4–5 µm; up & down IR flux: 4–50 µm; $\lambda/\Delta\lambda$ ~ 0.1–100	Optical window	
				H ortho/para ratio at P>3 bar	Acoustic Composition Experiment	TBD		

Table A- 2: Nominal Descent Duration 90 Minutes

Science theme	Science Priority	Science Objective	Scientific Measurement requirements		Instrument	Instrument requirements	Mission requirements	Comment
			Physical Parameters	Observables				
TRITON	1	Map surface geology at the landing site	Local high resolution surface geomorphology.	Visible reflectivity	Visible Imager: - Descent / down looking camera - Panoramic stereo camera - Microscope	Panoramic, stereo views. Images at different spatial scale: from 100s m down to few microns		
	1	In situ surface and subsurface characterization	Mechanical, electrical and thermal properties of surface, soil and subsoil	Surface physical properties at landing site: e.g. material density, cohesion, porosity, hardness, permittivity, thermal inertia, soil and subsoil temperature	In situ science package including e.g. - penetrometer - mechanical & electrical sensors - temperature sensors & heaters			
	2		Presence of a conductive layer or dynamo	Induced/intrinsic magnetic field, direction and magnitude	Magnetometer			*In situ ground truth for S/C observations*
	2	Determine surface composition, including organics; search for variations evidence for mass exchange/volatile transportation	Surface composition and microphysical properties of soil	Surface composition	Visible/Near IR spectral imager	Wavelength: visible to ~4 μm;		
			Geologic context and compositional variations	Surface morphology and microphysical properties of soil	Visible wavelength imager	Local high-resolution imaging panchromatic as a minimum, stereo views		
			Search for variability at surface and interaction surface/atmosphere	Sublimation / condensation cycle, volatiles, dust activity	Visible imager Gas analyser/MS Dust sensor	Observations at - different spatial scales (m-μm); - different local time		

Science theme	Science Priority	Science Objective	Scientific Measurement requirements		Instrument	Instrument requirements	Mission requirements	Comment
			Physical Parameters	Observables				
TRITON	2	Determine the composition of Triton's atmosphere	Atmospheric composition	Atmospheric species (N_2, CH_4, CO and CO_2) noble gas and isotopic abundance (C, Ne, Ar, Kr, Xe), complex organic molecules	Gas Analyser Mass Spectrometer (MS)	Mass range 1-300 amu		
	3	Investigate moon-magnetosphere interactions	Magnetic and plasma environment	Magnetic field direction and magnitude Ion and electron energy spectra vs time	Magnetometer Plasma and energetic particle detector			*In situ ground truth for S/C observations*

Table A- 3: Triton Lander Traceability Matrix